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Operational Loads Data

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ADVISORY GROUP FOR AEROSPACE RESEARCH AND DI VELOPMENT

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OPERATIONAL LOADS DATA

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PREFACE

This Specialists Meeting provided an opportunity for a useful exchange of views and a comparison of differing approaches to operational loads data acquisition. Such data acquisition was seen as a vital activity, needed to verify or update data available at the design and development flight test stage which, unsubstantiated by service experience, might lead to major inaccuracies in fatigue or static strength calculations.

There was general agreement that microprocessor-based systems provided the necessary technology for major advances in loads data acquisition techniques. Several such systems are under active development, others in routine use. In general, those in service appeared to assume the retention of correlations between flight parameters and loads data acquired in flight testing, whereas some newer systems placed greater reliance upon direct load or stress (strain) measurements. In either case, there was general observance of the '2-tier' concept, whereby a small proportion of an aircraft fleet is instrumented for extensive loads data acquisition, the remaining aircraft being monitored by simpler methods.

The main emphasis of the papers delivered tended to fall on data acquisition for fatigue strength assessment and fatigue testing, although it was intended that synthesis of data for use in design load calculations should also be adequately covered. Some papers described the collection of loads data from standardised test manoeuvres to define maximum design or fatigue loads accurately, although several participants felt that service pilots' handling was too variable for this approach to be entirely reliable. Discussion also queried the applicability of data from conventionally-configured aircraft of the present generation to some new designs, e.g. highly agile, unconventional lay-outs, largely constructed of composite materials and heavily dependent upon active control technology. The indications were that operational loads data acquisition would become even more vital in the future than for current military aircraft.

'Design Load Requirements' and 'Combat-NATO-Manoeuvre Flight Tests' are possible subjects for follow-on activity within the Panel.

D.M.F.BRIGHT
Chairman, Sub-Committee on
Operational Loads Data

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INTRODUCTORY REMARKS

by
Wing Commander D M F Bright RAF
Ministry of Defence (Air)
Whitehall, London, UK

The acquisition and use of operational loads data is a topic which has been periodically addressed by the Panel. Since it has not been reviewed within the last 5 years, I was one of those who suggested a need for renewed discussion and this Specialists Meeting, taking into account recent relevant experience within the NATO member nations. The subject appears a popular one, judging by the attendance at the Meeting. We will have the opportunity to listen to many interesting papers, and contrast and compare their various viewpoints and approaches.

I propose to open the proceedings by expressing a personal viewpoint, reflecting experience within one user service, namely the Royal Air Force. Our current fleet is a wide mixture of aircraft types which fall broadly into 2 categories: a number of older types, which have been retained in service for periods much longer than was originally anticipated; and a group of newer designs which I can characterise as high-performance 'fast jets'. Both groups have suffered from a number of significant structural defects, which have at times affected operations, due to inspections, long rectification down-times and some constraints imposed on flying; many structures have also been costly to repair. We are, of course, not unique in this respect; most other military operators have had similar experiences.

Almost without exception, these structural defects have not been caused by accidental damage nor, directly at least, by operational factors, including overload; nor are they attributable to initial defects on build, ie.poor quality control. On the contrary, they almost invariably stem from fatigue damage in metallic structure, generated by the effects of repetitive operational loading. Either the true stress levels were later found to be higher than anticipated, the frequency of damaging load cycles was greater than expected, or the fatigue resistance of the local structure had been over-estimated. At a detail level, these untoward events are sometimes due to stress concentrations being sharper than design stage analysis would suggest or a complex redundant structure reacting to loads in a manner which the designer did not foresee and allow for. However, if flight and ground loads are accurately known, there seems no fundamental reason, given the power of modern stress analysis techniques and sound design practice, why such situations should be repeated in the future. On the other hand, in many instances loads spectra can be substantially more severe than were ever expected, in which case the validity of the whole fatigue or strength analysis is inevitably undermined at the very outset.

As users, we therefore have a major interest in ensuring that accurate operational loads data is fed back to the designer. The service lives of military aircraft are frequently prolonged, many aircraft are pressed into new roles, new equipment is often fitted, and all-up weights and installed power levels generally increase markedly over the years. Operational demands also tend to become more severe, even if roles do not change - examples are the increased intensity of flying at low altitude,the introduction of new combat manoeuvres and reliance upon air-to-air refuelling to extend range or loiter time. We may not always fully appreciate the structural effects of some routine manoeuvres or variations in handling, the significance of severe discrete gusts at low level, or the fatigue implications of flight control system design characteristics. For all these reasons, we need better data on the loads imposed by normal service, to maintain and assure long-term structural airworthiness.

Looking to the future, active control technology is clearly going to have a major impact on military aircraft design. User pressure for greater structural efficiency is going to increase, via demands for maximum performance. We see the rapid introduction of advanced composite structures, which will almost certainly alter the current balance between fatigue and static strength requirements. Finally, next-generation aircraft will be designed to operate within wider flight envelopes, with increased use of unconventional manoeuvre capability and configuration control. The need for high-quality operational loads data will therefore increase, not decrease, in the coming years.

This meeting provides us with an opportunity to discuss the acquisition and exploitation of operational loads data, the choice of system philosophy, practical constraints on system design, and - last, but not least - the requirements for minimum cost, complexity and impact on the user, who has to fly the system and provide the data. I am confident that we will all learn a great deal from the proceedings. Discussion periods have been inserted into each session to allow full discussion of the papers given and an exchange of views on the topics raised.

OPERATIONAL LOADS MEASUREMENT: A PHILOSOPHY AND ITS IMPLEMENTATION

by

Dorothy M. Holford and J.R. Sturgeon
Materials and Structures Department
Royal Aircraft Establishment
Farnborough
Hampshire
GU14 6TD
United Kingdom

SUMMARY

A philosophy of operational data acquisition, for structural objectives, within the general field of in-flight load measurement is reviewed, highlighting the constraints such activities place on the data acquisition system. This Report describes one such system which can be tailored to perform a variety of tasks ranging from the collection of time histories of flight parameters or strain gauges to complex fatigue load analyses throughout the airframe. The system comprises a digital cassette recorder and a data acquisition unit within which a microprocessor is used for control of data acquisition and in-flight data analysis. System requirements in terms of accuracy, bandwidth and sampling rates are discussed for a range of aircraft types and operating conditions.

The various modes of operation of the system are illustrated by examples drawn from operational experience with the system. These demonstrate the capability of the system to produce data suitable for automatic analysis in a variety of operational environments in both fixed and rotary wing aircraft. The examples clearly show the value of studying operational data in terms of fatigue life management, fatigue life monitoring, operational practices and design procedures.

1 INTRODUCTION

Structural repair and maintenance costs can be several times the original cost of aircraft production. There is therefore, potentially, a rich reward for effective structural management within a fleet. The UK has directed its operational flight data acquisition and associated research programmes towards reducing unscheduled repair costs through a validation of the fatigue substantiation process, through understanding the causes of high fatigue damage rates and through the development of effective monitoring of structural usage fleetwide. Knowledge of the causes of high fatigue damage rates permits identification of load alleviation strategies which have no operational penalty. It is considered that realization of the first two aims can be accomplished by a comprehensive instrumentation fit to a relatively small number of aircraft covering the roles and theatres of the fleet. The aircrew operating procedures for the sample aircraft must be identical to other aircraft in the fleet so that they are not singled out for especial treatment. The investigative nature of the tasks demands that any airborne processing of the data shall not compromise the structural usage of the data, or eliminate the capability to diagnose faulty data.

Most UK military aircraft carry a fatigue load meter - a counting accelerometer with special levels and thresholds - the output of which can be used to monitor the fatigue life consumption of those components of the aircraft whose loading has a high correlation with normal acceleration in all fatigue damaging situations. To interpret the exceedance counts, assumptions must be made about the load distribution over the aircraft at the time the counts were registered. It can be seen that, for the modern combat aircraft with automatic stability and control systems, manoeuvre devices and other configuration control, use of the fatigue meter leaves many components unmonitored and gives rise to considerable uncertainties in the calculated fatigue consumption in others. Data from the operational load measurement programme can be used to improve the efficacy of current monitoring techniques and, if necessary, determine suitable alternatives.

In the course of the programme outlined above, methodology, airborne recording equipment, ground replay and analysis procedures have been developed and tailored to meet the exacting demands of the programme. It has been a process of evolution with successive programmes growing in complexity and benefiting from experience gained from preceding ones. All operational load measurement programmes are financed and monitored technically by MOD, the analysis of the data being carried out by the appropriate division of the aircraft contractor. This Report describes the overall philosophy of the load measurement programmes, the development of the recording system used in the majority of the programmes and illustrates its diverse capabilities with examples drawn from operational experience with the equipment.

Although this Report concentrates on the fatigue load measurement and monitoring aspects of data analysis, it is pertinent to note that the data are also examined to assess the adequacy of aircraft design requirements. The operational load measurement programme can also yield information of use in the design process. The nature of the data analysis required in this context is the subject of further research programmes since the perceived nature of the structural loading and, to a certain extent, the

aircraft response parameters are a function of the collecting aircraft's characteristics. These characteristics must be eliminated from the data in order for them to be relevant to a future aircraft.

2 DATA COLLECTION PHILOSOPHY

2.1 Operational load measurement programmes

These programmes seek to quantify the in-service fatigue life consumption and to determine the cause of high fatigue damage rates. The validation of the fatigue substantiation process invariably involves deduction of stress histories at particular structural features during service operation which can be assessed against those used in the determination of the fatigue performance of those features, eg as measured on the major fatigue test. The operational programme must collect loading data in a form which is most directly usable. Each component or major load path addressed by the programme must be considered from this standpoint. Essentially the question being asked is - what loading information is required given the fatigue substantiation process used? It is vital to pose this question in the planning stage because the analysis task must be defined and its implementation to hand when the operational data arrives. Critical analysis must proceed as the data are acquired to detect faults and false assumptions quickly. Otherwise the data will be consigned to storage and never be used. The fatigue analyst usually requires structural data in the form of local stresses or overall applied loads - torque, bending moment and shear - at particular stations. The operational load measurement programme must provide these data.

In the UK, consideration was given to evaluating structural loads from flight control and response parameters together with aircraft configuration and flight condition data. There are two distinct ways of dealing with the parametric data and it is important to be aware of the limitations inherent in each when used to calculate structural loads for fatigue purposes. In the first it is necessary to make assumptions about the underlying aerodynamic and inertial load distributions giving rise to the parametric data obtained. This must be done throughout the whole flight envelope for the many operating and environmental conditions met in practice. These load distributions are frequently not validated for each aircraft component. At frequencies appropriate to the aircraft's rigid body modes the totality of aircraft loading derivatives will have been used in handling investigations and in studies of flight control law performance etc. It is quite possible that the set of derivatives will have been tweaked to reproduce certain aircraft characteristics. However, such models frequently utilize parameters which are difficult to measure accurately. Additionally asymmetric flow breakdown, turbulence, aerodynamic interference effects and structural vibration all influence the required load distribution and are not necessarily reflected accurately in the whole aircraft model. Identification of these conditions and their effects on the desired load time histories further compound the interpretation of the parametric data in respect of applied load distributions. Once the load distribution has been found, overall loads can be calculated or local stresses found from a structural model (eg a finite element model) - a process not without difficulty!

Alternatively, use may be made of load prediction equations[1] which are derived from a data base of loads and parameters by regression techniques[2]. These load prediction equations often reflect the statistical properties of the data base and may or may not represent the underlying physics of load generation. Their use outside the domain of the data base is therefore not advisable. Consequently the data base used to derive the parametric equations must be carefully selected, having regard to the number of specific manoeuvres/flight conditions occurring in service and their contribution to fatigue damage. This is unknown at the outset of the operational load measurement programme. Non-linearities and strong dependence on flight condition and vehicle configuration complicate and increase the number of load prediction equations required.

There are clearly many difficulties and uncertainties in evaluating structural loads from parametric data. Since the objective of the operational load measurement programme is to quantify the operational fatigue loading, the UK has favoured direct load measurement using strain gauge installations in those programmes. Load histories derived from strain gauge installations automatically include compensation for configuration changes and operational procedures whereas those derived from parameters do not. However, calculating loads from parameters means that the analyst can, in theory, compute loads anywhere in the structure whereas with direct load measurement he has to make a choice. His aim should be to quantify the loading conditions in the main load paths to give overall confidence in the fatigue substantiation procedures and in any known local fatigue sensitive areas. The gauge installation itself has to be carefully engineered to survive in the operational environment and produce usable data over long periods of time. Erratic performance of potentiometers, plugs and sockets, variability in insulation resistance, sensitivity to temperature and electrical and radio frequency interference must be eliminated, as far as possible, by careful design. All strain gauges are wired as complete four arm bridges at the measurement position. Where possible, strain gauge amplifiers should be sited close to the gauge installations so that long cable runs carrying low voltage signals are avoided.

The location and orientation of the strain gauge is critical since it gives a measurement of direct structural strain along its length in the immediate area of its location. In operational load measurement programmes gauges should be attached to well defined load carrying elements and in uniform and unidirectional stress fields away from stress concentrations. Each gauge installation should be chosen so that it

predominantly responds to a desired single loading action unless the critical fatigue location is sensitive to more than one loading action. Local strain measurements can be used directly in fatigue calculations but, more usually, a 'fatigue calibration' is effected through a similar gauge installation on a major fatigue test specimen because it would never be wise to put an operational strain gauge at the local stress concentration which initiates the fatigue failure. Comparative spectra and rates of accumulation of fatigue damage can thus be established between the test specimen and the operational aircraft. Sometimes it is necessary, because of the fatigue substantiation process used, to establish the proportions of component stresses at a critical feature or to estimate the overall applied torque, bending moment and shear distributions. In these circumstances a 'load calibration' is required. The latter is mainly used where it is necessary to relate the operational measurements to the input of a major fatigue test or, if sufficient detail can be provided in the overall load estimates, to permit fatigue calculations to be performed over the whole structure.

The structural loading data can be analysed for fatigue but to elucidate the causes of high fatigue damage rates the analyst must be able to establish associated aircraft motions and automatic system performance. Thus, in addition, flight parameters detailing flight condition, aircraft configuration, control demands and aircraft responses are also measured. Where appropriate the performance of stability augmentation systems and the like are also monitored. The parameter list is tailored to complement the structural fit and varies from aircraft programme to aircraft programme.

2.2 Fatigue monitoring

A fleetwide monitoring system should provide estimates of fatigue damage for critical structural components from the measured structural usage of the aircraft; give an indication as to whether a structural inspection is necessary; provide sufficient supplementary information to enable the operator to put a structural cost on his operations; and provide an assessment of the loading environment in major load paths and components for comparison with design assumptions.

For fleetwide implementation, the ground processing costs must be contained and disruption of operational turn-round procedures minimized. The results must be examined quickly in order to check serviceability of the monitoring system and identify damaging sorties. The cost and ease with which monitoring information can be used will dominate the issue of acceptance and usefulness of the system to the operator. This means that in the monitoring system the majority of the data processing must be done in real time during the mission. The fatigue meter is an excellent example of this philosophy. Its successor must provide more detailed coverage of the airframe and improve the accuracy of fatigue estimates by collecting structurally relevant statistics of load histories appropriate to each component monitored.

The need for more comprehensive fatigue monitoring on modern combat aircraft[1] is illustrated by the calculated taileron loading during two stylized rolling manoeuvres shown in Fig 1a&b. The normal acceleration is comparable in both cases but there is a factor of 2½ between the associated taileron loads which represents a factor of 15 to 50 on fatigue damage. Fig 1c illustrates the effect of manoeuvre demand control systems where, in the case illustrated, a rudder kick gives rise to taileron loads but no normal acceleration increment.

In the context of a fleetwide fit, there is obvious attraction in computing structural load time histories from easily measurable flight parameters and each case must be examined on its merits. For the case illustrated in Fig 1, acceptable load estimates can be obtained from a linear combination of normal acceleration at the centre of gravity, roll rate, symmetric taileron angle and differential taileron angle for a given flight condition, aircraft configuration and low angle of attack[1]. However the physcial loading mechanism in this flight regime varies markedly with aircraft configuration and flight condition. At high subsonic Mach numbers, different parametric equations are required for clean and multiple stores configurations. Also each equation is applicable over a restricted range of flight conditions. A large number of load equations will be needed to cover all flight regimes. This will render the parametric approach untenable if the operational load measurement programme shows that substantial fatigue damage is sustained in many flight regimes. By way of contrast, early work by Anne Burns et al[3] suggested that quite simple equations could be used to predict fin loads over a wide range of flight conditions. However the data[3] used did not contain significant information at frequencies above 1 Hz because low pass filters were used to remove activity at structural frequencies. Section 6.2 indicates that this bandwidth limitation leads to gross underestimates of damage rates in some flight conditions.

In a fleetwide monitoring system only those statistics of the loading environment necessary for fatigue evaluation can be economically computed. Currently it is envisaged that complete loading cycles within a loading waveform will be identified by a range-mean-pairs (rainflow) technique[4,5], and subsequently used with the remaining peaks and troughs to estimate fatigue damage (or crack growth) using simple cumulative damage algorithms. If necessary, residual stress effects (or crack retardation) can be introduced on a flight-by-flight basis.

An analysis technique to meet the objectives of fatigue monitoring is described in Ref 6. It provides damage estimates at regular intervals during the flight based on interim estimates of the fatigue resistance of the structural components. To find the causes for high fatigue damage rates it is necessary to supplement these estimates with

broad brush aircraft state/configuration parameters at the time of the damage estimates. Such parameters might include the range of altitude, airspeed, normal acceleration at cg, lateral acceleration at cg, flap, slat, airbrake, nozzle angle, sweep angle etc during the last time interval. These data give the operator the basic information for fatigue management of the fleet. The technique is illustrated in Fig 2 which shows the fatigue damage profile for the wing of a large flexible aircraft. (The data are taken from Ref 6.) High fatigue damage rates and the relevant operating conditions can be identified and their validity established, to a certain extent, from the supplementary parametric information.

Load time histories directly related to fatigue critical features may be processed in the air to give, for each monitored station, fatigue damage on a flight-by-flight basis and a fatigue damage accumulation profile within a flight to meet the immediate objectives of fatigue monitoring. However the results of the intermediate process of loading cycle identification, ie the range-mean occurrence matrix for the whole flight, must be retained for future reassessment in the light of increased knowledge of the fatigue resistance of the structure.

3 RAMIFICATIONS OF DATA ACQUISITION

Studies of the structural implications of operational practices demand very high data quality for the analysis to be meaningful - complete flights must be analysed automatically and any data losses must be uncorrelated with structural loading severity. In the past many recorders have functioned reliably during innocuous flight conditions and consistently failed to function satisfactorily during structurally or aerodynamically severe flight conditions[7]. In operational load measurement programmes described in section 2.1, the data must be recovered in time history format to permit detailed investigation of the causes of high fatigue loads. The performance of the data acquisition system can therefore be readily assessed.

Of equal importance is the performance of the sensors. For the operational load measurement programmes to succeed, sensors must survive in the operational environment. Many data collection programmes have collected flight response data from accelerometers, potentiometers etc. These experiences suggested that these sensors would function reliably in the operational environment. There was however a question mark over the durability and stability of strain gauge installations in the operational environment. The first UK programme of the type described in section 2.1 began collecting data in 1977. Thirteen strain gauge bridges were attached to the wing of a Victor tanker aircraft. Of these one gauge failed within the first three months but the rest remained serviceable throughout the programme which collected data for three years. Gauge datums and amplifier sensitivities remained sensibly constant throughout the programme. This encouraging experience sugested that, with placement of spare gauges alongside those to be recorded, the data acquisition programmes could produce structural loading data from strain gauges in service. However unserviceable sensors must be detected quickly otherwise much useless data may be collected. This means that the data must be verified for quality as soon as possible after collection, preferably within a week.

In the Victor programme mentioned above data were recorded on a fourteen track analogue FM recorder. Thirteen of the fourteen tracks were given over to strain gauge data while the flight response data were sampled digitally and written to the fourteenth track. The analogue recorder was used in our first programme for expediency. Digital data acquisition systems are preferred since they offer greater flexibility with respect to the number of parameters sampled and are less susceptible to electrical interference. However the Victor programme did give an opportunity to study the effects of limited bandwidth and sampling rates on fatigue loading patterns. These and other studies (sections 6.2 and 6.3) suggest that only the lower order structural modes are likely to be fatigue damaging in their own right. The major loading cycles emanate from control surface motions and the atmospheric environment but their magnitude may be substantially enhanced by the superposition of vibration loads. This means that typically load histories must be sampled at 8 to 64 samples/s depending on the predominant structural resonant frequencies. This provides information on the fatigue implications of the vibration but may not enable its frequency and phase characteristics to be deduced.

In order to mount an operational loads measurement programme on a small combat aircraft, the data acquisition system must be relatively compact. However with compactness comes comparatively short duration recording at the required sampling rates. Therefore timewise data compression techniques must be used to reduce the quantity of data collected during structurally innocuous flight. This requires 'intelligence' on the part of the data acquisition system which can be supplied through a microprocessor based system. It is worth remembering that ground validation of the data and fatigue analysis are time-consuming and expensive and are proportional to the number of data words collected. Therefore the use of data compression techniques prior to data recording leads to a cost effective usage of ground analysis resources.

As discussed in section 2.1, for the fleetwide fatigue monitoring system the bulk of the fatigue analysis must be accomplished in the air. If, as is likely, the fatigue consumption of the modern combat aircraft can only be effectively monitored through access to a subset of the parameter list of the operational load measurement programme then it is clearly logical to think in terms of common equipments.

Engine Usage Monitoring Systems (EUMS) have been under development, under the direction of the Directorate of Engines MOD(PE) since the early 1970s[8]. The data

acquisition and recording system was developed by Plessey and uses a Davall compact cassette tape recorder. The EUMS Mk I had an overall sampling rate of 32 samples/s and recorded data continuously on a single track of the cassette tape. There was no analysis capability. The EUMS Mk II was conceived as a microprocessor based system, with higher sampling rates and some airborne computational capacity. Materials and Structures Department, RAE have participated in the development programme with the aim of producing a versatile data acquisition and analysis system which would meet the diverse tasks of the operational loads measurement programmes.

Utilization of common data acquisition hardware for both structural and engine loads data measurement maximizes the return for equipment development costs. Scant research and support service resources are thus directed towards a definitive single entity with correspondingly greater return for that effort.

4 BASIC FEATURES OF THE EUMS MK II DATA ACQUISITION SYSTEM

The airborne data acquisition system that has evolved[9] is based on a digital flight recorder with microprocessor controlled data management. The microprocessor is used to control the acquisition of the data, its subsequent intermediate storage in a solid state memory and its final blocked transfer to compact cassette via a Davall 1207-003 tape recorder. The hardware development was carried out by Plessey under MOD contract. The main components of the airborne system are shown in Fig 3 in bench test arrangement. The system comprises the data acquisition unit, control unit and quick access tape recorder. The data acquisition unit accepts analogue signals from all standard transducers and converts them to 10 bit numbers at the desired sampling rates. Parameter sampling rates are controlled by software as is their position in the data frame format. The overall sampling rate is switchable and ranges from 8 to 512 words per second in binary steps. Thus common digital recording hardware may be tailored by software changes to meet specific requirements. Signal conditioning boards can also be configured to suit particular applications[10].

The 10 data bits, an eleventh bit (used as a mode marker for timewise data compression, section 7) and a parity bit are assembled into a 12 bit word and stored in solid state intermediate memory silos. The data stream is assembled in an ARINC 573 code pattern[11]. Software controls whether or not a silo of data is subsequently transferred to cassette. The tape drive works in a stop/start mode and if data are to be output to tape, the tape is run up to its operating speed of 3 in/s when data are transferred from a silo of 2560 words at 1540 bits/in. When the silo is empty the tape drive stops. This recording mechanism results in a substantial increase in reliability since the tape transport mechanism is always recording data at a relatively high tape speed. Wow and flutter problems associated with low tape recording speeds are much reduced. Data quality is improved in the presence of severe structural vibration and the extreme event is certain to be actually recorded on the tape at least 1½ s after the event. By the use of parallel recording on four tracks of a compact cassette, over two million words are recorded on a single C-90 cassette. Data quality cassettes cost about £5 each and are thus an attractive data medium from an economic point of view. They are cheap enough to be used once only and also provide an economic means of long term data storage. In the absence of data compression the recording durations for a C-90 cassette range from 20 h at 32 words/s to 1½ h at 512 words/s. The compact cassette is quickly and easily transferred from its operational location to the ground analysis station. However there is also a family of ground support equipments which are used in the operational theatre to diagnose faults in the system and transducers. A flight line test set can be used to interrogate the digital data at selectable locations in the data format. Stability of ground datum values can be assessed for acceptability. A portable replay unit can produce hard copy data from the cassette if required.

5 OPERATIONAL EXPERIENCE WITH EUMS MK II

The pilot operational loads measurement programme on Victor clearly demonstrated the value of such programmes in removing speculation as to the origins and magnitudes of structural fatigue loading cycles. The first uses of EUMS Mk II in structural programmes on fixed wing aircraft were aimed at ad hoc problems and were restricted in their coverage of the airframe and the associated parameter fit. The programmes utilized equipments procured for engine usage studies, as in the case of Jaguar, or research purposes (Hercules and Sea King). The programmes are outlined in Table 1. The fitment of the system in a Sea King helicopter was by way of a pilot exercise to gather both flight usage data and engine torque data to establish the viability of the data acquisition system in the helicopter environment.

At RAE, data from each installation were studied in parallel. A number of sources of data corruption in the complete data acquisition/ground replay system were identified[12]. During these early programmes there was a continuing improvement in the quality of the data as successive error patterns were nullified. The installation on Jet Provost in late 1981 incorporated most of the system improvements identified in the earlier programmes and discard rates of 1 in 10^4 to 10^5 words were achieved.

Residual errors were not associated with structural severity. It is likely that most were associated with power supply transients. Future installations will include a transient suppression unit which will hold up the power during a 50 ms break and also remove large spikes giving a stabilised power supply. These programmes have demonstrated that the system can produce data of high quality in the operational environment. However it must be recognized that no airborne digital recording system will ever

produce flawless data. The analyst must ensure that error patterns in the data stream do not invalidate his analysis.

For the large transport aircraft, the sortie time is likely to exceed the cassette recording time. The simple EUMS Mk II becomes less attractive since crew action is required to change cassettes. However, with the Hercules programme, cassettes were changed at regular intervals enabling data to be collected throughout the sortie. It does mean that there are special instructions for operating the data collection aircraft of the fleet. This type of notoriety is usually avoided in an effort to ensure the data collected are representative of fleet usage. For future programmes, utilization of the EUMS Mk II with timewise data compression (section 7) may well obviate the need for crew action.

6 EXAMPLES TAKEN FROM THE OPERATIONAL LOAD MEASUREMENT PROGRAMMES USING EUMS MK II

6.1 General remarks

As mentioned in the Introduction, the operational loads data are analysed by the aircraft contractor to meet the direct objectives of the exercise. The examples presented here are taken from the investigative and research programmes at RAE. They illustrate that EUMS Mk II works well in diverse environments. The data have, in the main, been used to study the impact of sensor characteristics on analysis tasks. This work enables bandwidth and sampling rates for the more ambitious programmes (section 7) to be estimated with confidence.

The examples clearly show the need for accurate fatigue monitoring and illustrate the difficulties that are associated with deriving loads from aircraft response parameters. The strain gauge installations remained serviceable throughout the duration of the programmes (Table 1).

6.2 Empennage load measurement

6.2.1 Jaguar

The Jaguar programme showed that low-level operations were a major contribution to fatigue damage at the fin root. It was therefore most important that subsequent programmes should accurately assess the situation on other aircraft. Of the two aircraft in the Jaguar programme, one had a strain gauge amplifier bandwidth of 10 Hz and the other 40 Hz. The fundamental fin bending mode is about 12-13 Hz. Visual inspection of data from the 40 Hz bandwidth aircraft suggests that a few high frequency loading actions are eliminated by the 10 Hz bandwidth amplifiers. The following discussion is based on data from the 10 Hz bandwidth system sampled at 32 samples/s. The life consumption analyses can be considered reasonably realistic for the real structure but are related to specific flight conditions. Any read across to total aircraft lives must include an assessment of how often the flight condition is met in practice.

Fig 4 shows a fatigue damage profile for a low-level sortie. The rate of fin fatigue damage varies within a patch of low-level flight, the damage rate quickly falling to zero (over a period of 5-15 s) when the aircraft gains altitude. The damage rate varied between flights - some flights had locally higher rates than those of Fig 4. These variations may well be associated with the terrain and/or the atmospheric environment. Strain data, typical of the period marked on Fig 4 are shown in the top trace of Fig 5 with an expanded time base to permit a more detailed study of its characteristics. The lower traces of Fig 5 show how these data are distorted as the bandwidth is reduced to 6, 1.8 and 0.9 Hz respectively by numerical filtering techniques. Dutch roll activity is a possible cause of the lower frequency component at about 0.8 Hz. The sharpness of the transient strain, perhaps due to a gust, at point A of the top trace of Fig 5 suggests that a higher bandwidth and sampling rate would be necessary to measure accurately its magnitude and characteristics.

As might be expected from a visual inspection of Fig 5, the bandwidth has a marked effect on computed life estimates and damage distributions. Damage rates per hour were estimated for the flight conditions of Fig 5 for each of the data bandwidths illustrated. A range-mean-pairs[4] analysis was used to produce the stress range exceedance count shown in Fig 6. It can be seen that relative to the raw data exceedance curve the 0.9 Hz bandwidth data reduces the stress range amplitude by a factor of 1.5 at the higher amplitudes and reduces the frequency of occurrence of the lower amplitude cycles by a factor of 3. The resulting damage distribution for the mean life S-N curve of Fig 6 is shown in Fig 7. The characteristic trend is for the maximum damage to be shifted to a lower stress amplitude and to be reduced in magnitude as the bandwidth of the data is reduced. The total damage is reduced by factors of 1.2, 2.8 and 12 for the 6 Hz, 1.8 Hz and 0.9 Hz bandwidths respectively.

6.2.2 Jet Provost

At entry into service the predicted fatigue life of the Jet Provost fin root was dominated by spinning. Unexpected fatigue failures led to an ad hoc operational load measurement programme. It was found that although the spins did produce severe structural activity, in fatigue damage terms, the type and extent of current low-level training proved more exacting. Based on Jaguar experience the strain gauge amplifiers of the Jet Provost programme had 40 Hz bandwidth, and fin and tailplane strain gauges were sampled at 32 samples/s.

Data from EUMS Mk II collected during stalls and spins are shown in Fig 8. It can be seen that the stalled wing condition leads to substantial high frequency structural activity at both the fin and the tailplane and some at the wing root. From the expanded time history of fin bending moment, Fig 9, it is evident that the raw data can only indicate the severity of the high frequency loading components during the stall condition; higher sampling rates and/or bandwidth are necessary for accurate assessment and to detect individual erroneous data points. The raw data can however be used to identify such flight conditions/manoeuvres for which a more detailed assessment is necessary should their frequency of occurrence in the operational spectrum indicate a significant contribution to the total fatigue life consumption. It is pertinent to note, from the two lower traces of Fig 9, that this is still evident when the sampling rate is halved to 16 samples/s.

The section of low-level flight shown in Fig 10 is taken from a 38 minute period of low flying and represents average amplitudes of gauge outputs during that period. The dominant oscillation on the fin bending gauge is at about 0.6 Hz and is probably associated with Dutch roll activity. The bursts of activity which grow and decay may well be initiated by the pilot and/or the atmospheric environment. In this flight condition significant structural activity is for the most part confined to the fin in contrast to the spin data above. A portion of fin gauge output was passed through the same set of numerical filters used in the Jaguar exercise. The result is shown in Fig 11. (The scales used are the same as those of Fig 9.) The 0.6 Hz stress reversals have superimposed on them varying amounts of high frequency activity which increase the fatigue damaging ranges. The increase is not uniform since the largest load cycle A of Fig 11 is virtually unaltered whereas others such as B and C are doubled when frequencies above 0.9 Hz are included.

6.2.3 General conclusions

Low-level operations have been a major contributor to fin fatigue problems on two aircraft: Jaguar and Jet Provost. In any assessment of fatigue damage throughout the operational spectrum it is important to use data of sufficient bandwidth and sampling rate so that fatigue damaging flight conditions/manoeuvres can be accurately identified and their damage contribution accurately quantified. It has been demonstrated, on Jaguar and Jet Provost operational flying, that empennage fatigue monitoring requires 10-40 Hz bandwidth and 32-64 samples/s. Only then can there be confidence in the calculated damage distribution and thus confidence in any read across to fleet lives.

6.3 Hercules

The prime objective of the Hercules programme was to measure wing loads during take off and landing. The only supportive flight parameter was normal acceleration. However data were collected throughout the flight. Some large sharp normal acceleration transients were seen on several flights during low-level support work. These were studied in some detail in view of the importance of low-level operations.

Current discrete gust requirements have evolved from normal acceleration data which were reduced to equivalent gust velocities under the assumption that the measured accelerations were due to the atmosphere. Normal acceleration data such as that of Ref 13 from the Civil Aircraft Airworthiness Data Recording Programme are frequently used in the study of gust models and statistics. However the analyst must ensure that the bandwidth of the raw data is adequate for his analysis objectives. The following example from the Hercules programme shows how limited bandwidth can mask the true characteristic of the rarer large sharp transients. The data obtained from the EUMS Mk II are shown in Fig 12. The local meteorological conditions were reported as 3/8 cumulus at 1800 feet with a mean wind speed of 17 kn. The transient at A has the characteristic of a rotor; the wing strain gauges exhibit a similar pattern. The derived statistics of the normal acceleration transient are shown in Table 2. These remain substantially unaltered as the bandwidth is reduced down to about 3 Hz. At 0.9 Hz bandwidth the character of the original transient has been destroyed, the amplitude having been reduced by a factor of 1.3 and the width doubled. Reducing the sampling rate further degrades the perceived properties of the transient. It is important to note that these data were obtained at 210 kn TAS. Many civil and military gust encounters are likely to be at higher speeds so higher bandwidth and sampling rates become essential.

The data of Fig 13 are taken from a combat training sortie at low level. The manoeuvre of the last 25 s of Fig 13 was repeated some 1½ min later but without the transient loading at A. Of particular interest is the transient at B, almost certainly due to gust since it is out of character for a deliberate manoeuvre. The maximum rate of change of g is some 7.5 g/s, the g increment being 1.5 g achieved over a period of 0.375 s. Both this transient and the previous one are comparable, in magnitude and gradient distance, with the discrete gust of design requirements. Unless the conditions giving rise to such transients can be identified by the pilot, then the simultaneous occurrence of the manoeuvre and the transient must be a definite possibility resulting in a structural load substantially exceeding current design requirements. Research programmes to study the structural risk of low-level operations are underway as are programmes to collect atmospheric turbulence data at low level over a variety of terrains.

6.4 Sea King

The data of Fig 14 were obtained from a EUMS Mk II installation in a Sea King helicopter and show a rapid turn from rearwards flight relative to the air and ground which was reported by the pilot. These data illustrate how difficult it can be to identify some manoeuvres and modes of flying from parametric data. The control angles shown are pilot demands: these actions together with inputs from the automatic flight control system give rise to the measured responses. It is difficult, if not impossible to identify the rearwards flight from these data. Other conditions such as sideways flight can be equally difficult to identify. If such 'unidentifiable' manoeuvres contribute markedly to the fatigue life of a component it will be difficult to assess that component for fatigue unless a direct measurement of the required loading is obtained.

The turn of Fig 14 has an average rate of about 36 deg/s; the pilot has no instrument to indicate its severity and his physiological cues indicate a fairly benign environment. Normal acceleration at the rotor station and the tail are very close to 1 g suggesting that the pilot station likewise is at 1 g conditions. The lateral acceleration at the cockpit is relatively innocuous but very large lateral accelerations and forces are generated at the tail particularly when stopping the turn.

Although the EUMS Mk II functioned competently in the helicopter environment the same cannot be said for one of the normal accelerometers at the rotor station which failed after about five flights. The trace, NCS, shown in Fig 14, is from the unserviceable instrument on the starboard side while NCP shows data from a serviceable instrument on the port side. The vibration at the accelerometer location produced a number of short circuited turns of the accelerometer potentiometer near the 1 g position. This produces a fault easy to recognize visually but difficult to identify automatically.

The pilot exercise on Sea King, which collected data from 150 flights, showed that the data could be interrogated automatically and analysed even though all airborne/ground system enhancements currently in use were not implemented. Forthcoming programmes on Chinook and Sea King will employ EUMS Mk II equipment.

7 TIMEWISE DATA COMPRESSION

In order to extend the recording capacity of the basic EUMS Mk II, software and hardware have been developed to implement a timewise data compression algorithm which does not impair the structural usefulness of the data. Data acquisition systems with this capability are known generically as Structural Usage Monitoring System (SUMS) recorders[14]. The incoming data are always sampled at a pre-selected maximum overall word rate, eg 512 or 256 words/s, and stored in an internal buffer of capacity 3 × 1024 words. The latest 1024 words are examined, on acquisition, for structural severity by reference to six user nominated 'trigger' parameters. Structurally significant flight is presently identified if any one of the 'triggers' exceeds a preset, programmable, limit. In principle any logical expression which is a function of the six 'trigger' parameters can be used. If structurally significant activity is deemed present then the previous 1024 words of data will be output at maximum rate otherwise only 64 (or a binary multiple thereof, eg 128 or 256) user-selectable words of the 1024 will be output. This method ensures that the output data stream contains at lea[illegible] s of pre-event data at high [illegible]mpling rate and accurately tracks slowly changing [illegible] conditions. Data are output at maximum rate until quiescent conditions have be[illegible]aintained for a given, user-programmable interval of time.

SUMS recorders will be used in forthcoming operational load measurement programmes on Tornado, Buccaneer, Hawk and Jaguar aircraft. The SUMS recorder for Tornado[14] will operate at a maximum rate of 512 words/s and will initially be used with a compression ratio of 8:1. In compressed mode, the 64 words in a second will contain a reading for each parameter taken from within a narrow time slice. It is expected that the flight time recorded will be extended from 1½ h to about 5 h. The first Tornado SUMS aircraft is expected to start collecting data in May/June 1984.

8 RANGE-MEAN-PAIRS ANALYSIS IN FLIGHT

During the fatigue analysis of the structural data from the operational loads measurement programmes, range-mean-pairs (rainflow) analyses are used to identify fatigue damaging loading cycles. The loading cycle is subsequently classified by its load range and mean load level and sorted into class cells whose boundaries reflect the requirement that each range-pair count in a cell can be assumed to produce a known amount of damage. The resolution used in the range-mean-pairs analysis must be sufficient to define accurately the boundaries of the matrix cells. The range-mean frequency of occurrence matrix is used in conjunction with a damage matrix to evaluate accrued damaged. Long-term storage of statistics of the loading environment, for future assessment, demands that only the structurally significant features of each load history be retained. The range-mean matrix provides the necessary basis (section 2.2). Clearly, producing the frequency of occurrence matrix in the air can reduce ground processing costs in the operational load measurement programmes if suitable load histories are defined a priori. Airborne analysis is a necessary prerequisite for the advanced fatigue monitor of section 2.2.

The microprocessor of the Hercules EUMS Mk II has been programmed to perform in-flight range-mean-pairs counting on all fifteen channels of data (14 strain gauges plus normal acceleration) in addition to its basic EUMS Mk II functions.

The occurrence matrices are accumulated in solid state memory and written to compact cassette at the end of the flight. The cassette therefore contains the raw data plus the results of in-flight analysis. Ground analysis has confirmed the correct functioning of the airborne computation.

The digitised loading data in a fatigue load monitoring system must encompass limit load conditions. At the same time the data must have sufficient resolution to cater for a variety of S-N curve shapes and low fatigue endurance limits (eg ±1000 lb/in^2, ±5.9 MN/m^2). To achieve this without introducing corrections for the grouping of the peak and trough values by the digitisation process, 256 resolution levels (8 bits) are needed. At 64 resolution levels substantial corrections for grouping will be required[5]. The EUMS Mk II has an 8 bit microprocessor and therefore the data were reduced from 10 bits to 8 bits prior to range-mean-pairs analysis.

A typical frequency of occurrence matrix recovered from the cassette tape is shown in Table 3. In this installation, the cell boundaries of the matrix were tailored to produce accurate estimates of damage due to low amplitude loading cycles postulating a low endurance structure. The cell sizes for the higher amplitude loading cycles are too large to allow exact calculation for any one flight because the counts per flight would be very low. However when many flights are summed, it is reasonable to assume that the counts, within a cell have a statistical distribution. The airborne algorithm[5,15] ensures that the largest amplitude cycle is separately identified as peak/trough values thus permitting a more detailed assessment of the effects of the once-per-flight loading cycle. Many more cells would contain counts during a severe flight.

9 CONCLUSIONS

The UK philosophy adopted in its operational load measurement programmes involves direct structural load measurement via strain gauge installations. These are supported by a flight parameter fit to identify the causes for high fatigue damage rates. Strain gauge installations have proved reliable in operational service.

The Engine Usage Monitoring System (EUMS Mk II), developed in the first instance for engine monitoring, has been used to collect the structural operational data on compact cassette. The system has been shown to work well in a number of different operational environments. The system is controlled by a microprocessor and can be tailored by software changes to meet specific requirements of a programme. The spare computing capacity of the microprocessor can be used for in-flight processing of the incoming data stream. In a particular application, range-mean-pairs analyses of fifteen data channels were performed and the results written to compact cassette at the end of the flight.

A derivative of EUMS Mk II known as Structural Usage Monitoring System (SUMS), has been developed to extend the flight time capacity of the compact cassette. In SUMS the microprocessor is used to effect a timewise data compression algorithm which does not impair the structural usefulness of the data. It is planned to use this data acquisition system for operational load measurement programmes on Tornado, Buccaneer, Hawk and Jaguar.

Data replay and analysis procedures have been developed to ensure that fault diagnosis and error correction can be controlled in the operational environment. The operational loads measurement programmes to date have, in virtually every instance, directed structural analysts to flight conditions/activities in which the structural penalties had not been fully appreciated. The programmes can provide a wealth of information on pilot operating techniques and the performance of automatic flight controls. This information can be used to produce design and operating recommendations that will conserve fatigue life with a negligible operational penalty. The programmes clearly demonstrate that different parts of the aircraft suffer fatigue damage in different operational activities and show that accurate fatigue life monitoring at many locations is essential to cost effective planning of fleet utilization.

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Table 1

EXPERIENCE WITH EUMS MK II IN OPERATIONAL LOADS MEASUREMENT PROGRAMMES

Aircraft	No.	Programme dates	Word discard rate	Parameters
Sea King	1	September 1980 - April 1981	3 in 1000	Flight response and control data, rotor rev/min and engine torques.
Jaguar	2	June 1979 - May 1980 June 1980 - September 1981	Data unusable 1 in 100	Three channels strain data on empennage, height and speed.
Hercules	1	December 1980 - April 1983	3 in 1000 (start) 1 in 10000	Fourteen channels strain data on wing, normal acceleration.
Jet Provost	2	October 1981 - April 1982	1 in 10000 to 100000	Eight channels strain data on empennage (7), and wing (1), normal and lateral (tail) acceleration.
Sea King	1	November 1983 -		Flight response and control data, rotor rev/min, engine torque, tail rotor torque, one strain gauge channel on airframe (choice of three switchable).

Table 2

EFFECT OF BANDWIDTH AND SAMPLING RATE ON THE CHARACTERISTICS OF THE HERCULES TRANSIENT 'A' OF FIG 12

Transient characteristic	Sampling rate 16 samples/s		Sampling rate 8 samples/s	
	Bandwidth of data		Bandwidth of data	
	15.5 Hz*	0.9 Hz	15.5 Hz	0.9 Hz
|g peak - g trough| (g units)	1.69	1.23	1.62-1.69	1.21-1.23
[illegible] peak - time trough| ([illegible]nds)	0.344	0.719	0.375	0.75
Δ[illegible]/Δt max (g units/s)	7.84	2.72	6.8-7.68	2.56-2.64

*The raw data has bandwidth 15.5 Hz and were sampled at 16 samples/s.

Table 3

HERCULES: RANGE-MEAN-PAIRS COUNT OF A WING STRAIN GAUGE

Matrix for Channel 13 HERCULES: RANGE-MEAN-PAIRS COUNT OF A WING STRAIN GAUGE

	Range in digits																					
	10-11	12-15	16-19	20-23	24-27	28-31	32-39	40-47	48-55	56-63	64-79	80-95	96-111	112-127	128-143	144-159	160-175	176-191	192-207	208-223	224-239	240-255
0- 31	0	0	0	0	0	0	0	0	0	0												
32- 63	0	0	0	0	0	0	0	0	0	0	0	0	0	0								
M 64- 95	17	22	9	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0				
E 96-127	1	4	0	1	0	0	0	0	0	0	0	1	0	0	0	0	0	0	0	0	0	0
A 128-159	76	81	25	7	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
N 160-191	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0				
192-223	0	0	0	0	0	0	0	0	0	0	0	0	0	0								
224-255	0	0	0	0	0	0	0	0	0	0												

Airborne manoeuvres and gusts

Possible rotation or bounce landing counts

Taxiway and runway counts

Ground-air-ground count

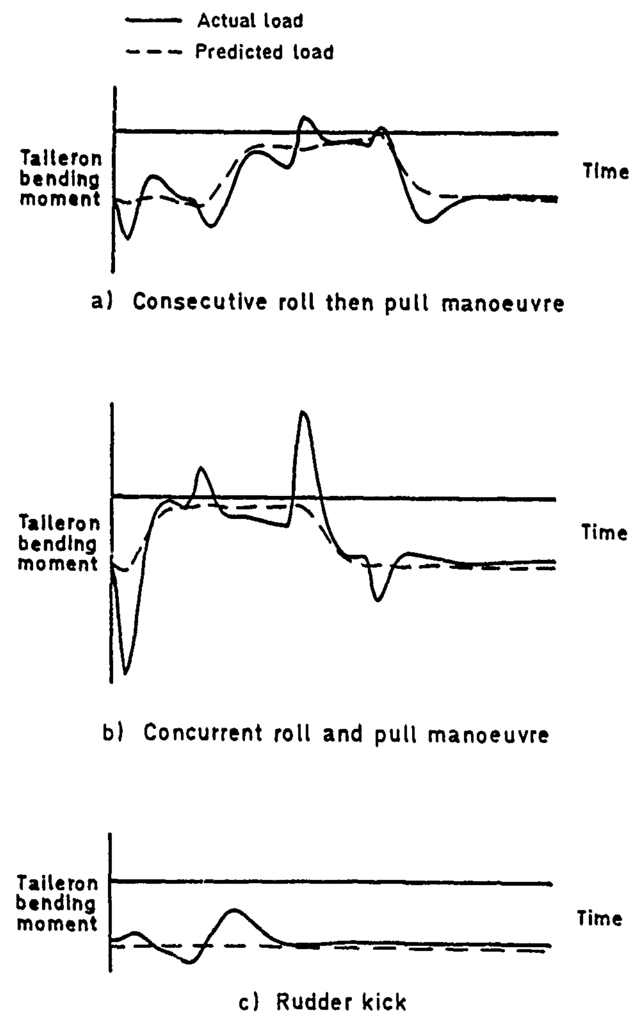


Fig 1a-c Comparison of actual load and that predicted from the normal acceleration at cg, n_z

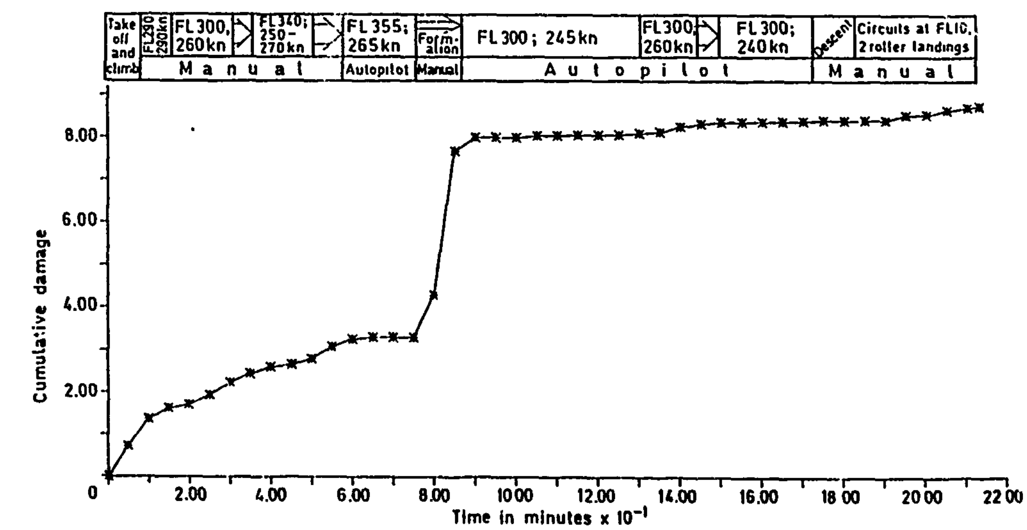


Fig 2 Damage accumulation profile and associated flight conditions for a wing feature of a large flexible aircraft

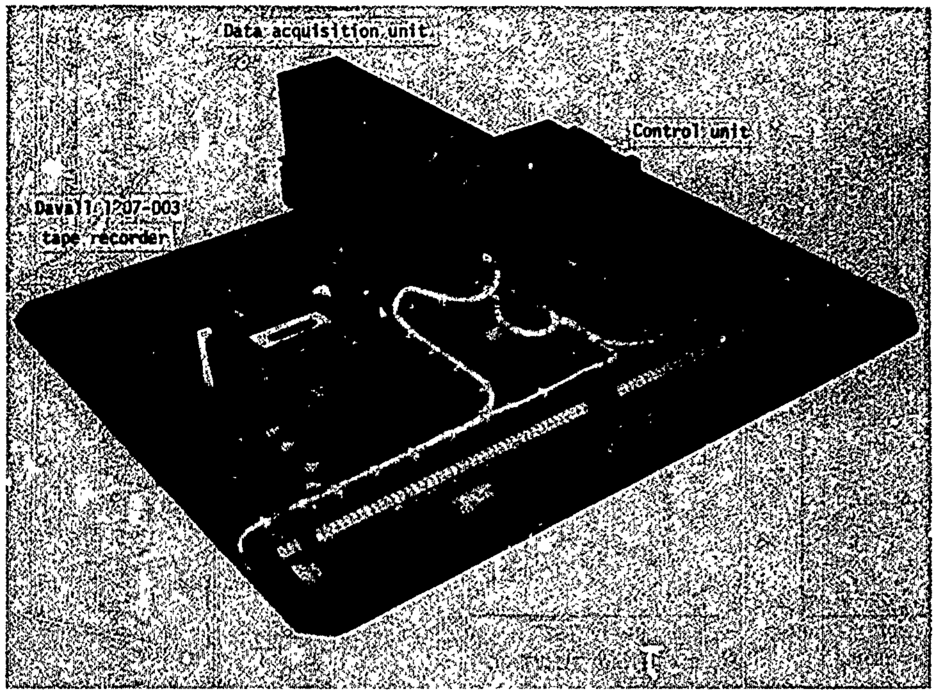


Fig 3 Operational load data acquisition system: bench test layout

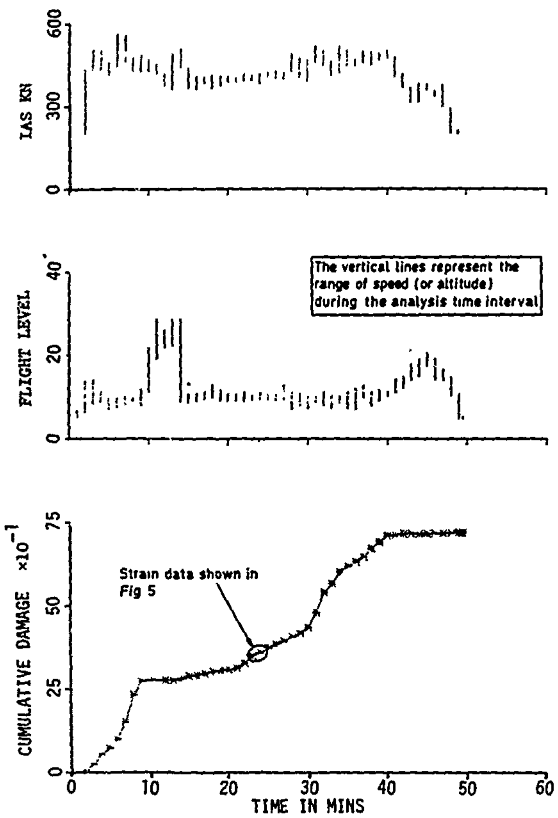


Fig 4 Damage profile and flight conditions for a low-level sortie

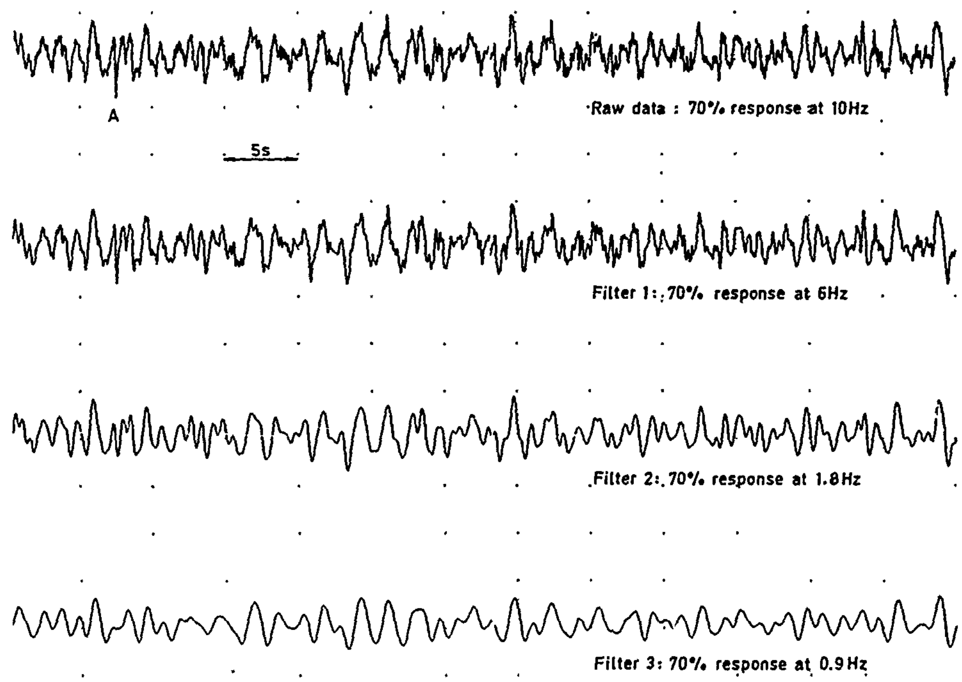


Fig 5 Effect of frequency bandwidth on Jaguar fin strain measurements during low-level flight

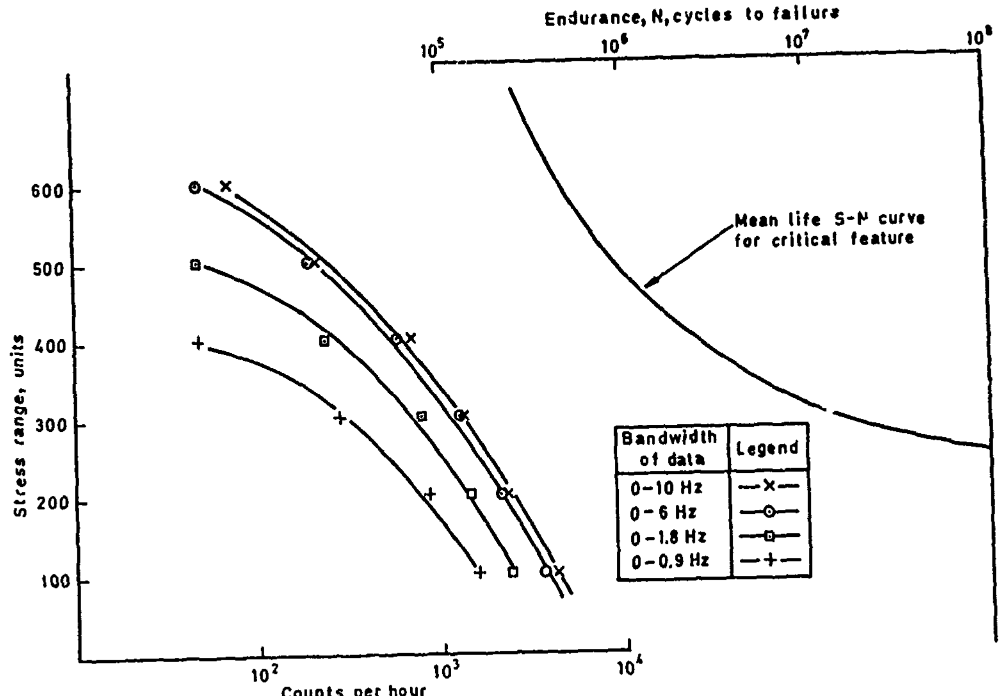


Fig 6 Effect of bandwidth on stress range exceedance count and S-N curve for a critical feature

The graphs join mid-cell points of the underlying damage histogram which is illustrated for +—+ curve only
The dashed lines indicate regions where the result might be unduly influenced by sample size

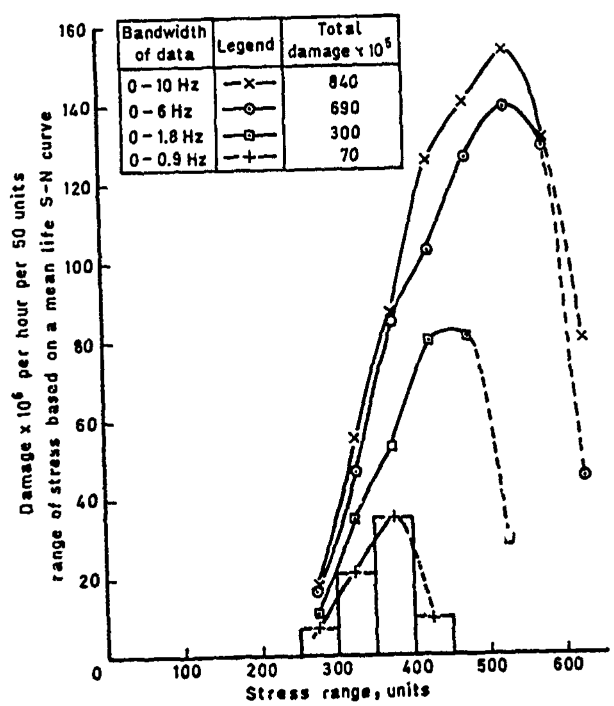


Fig 7 Effect of bandwidth on damage distribution for a particular feature

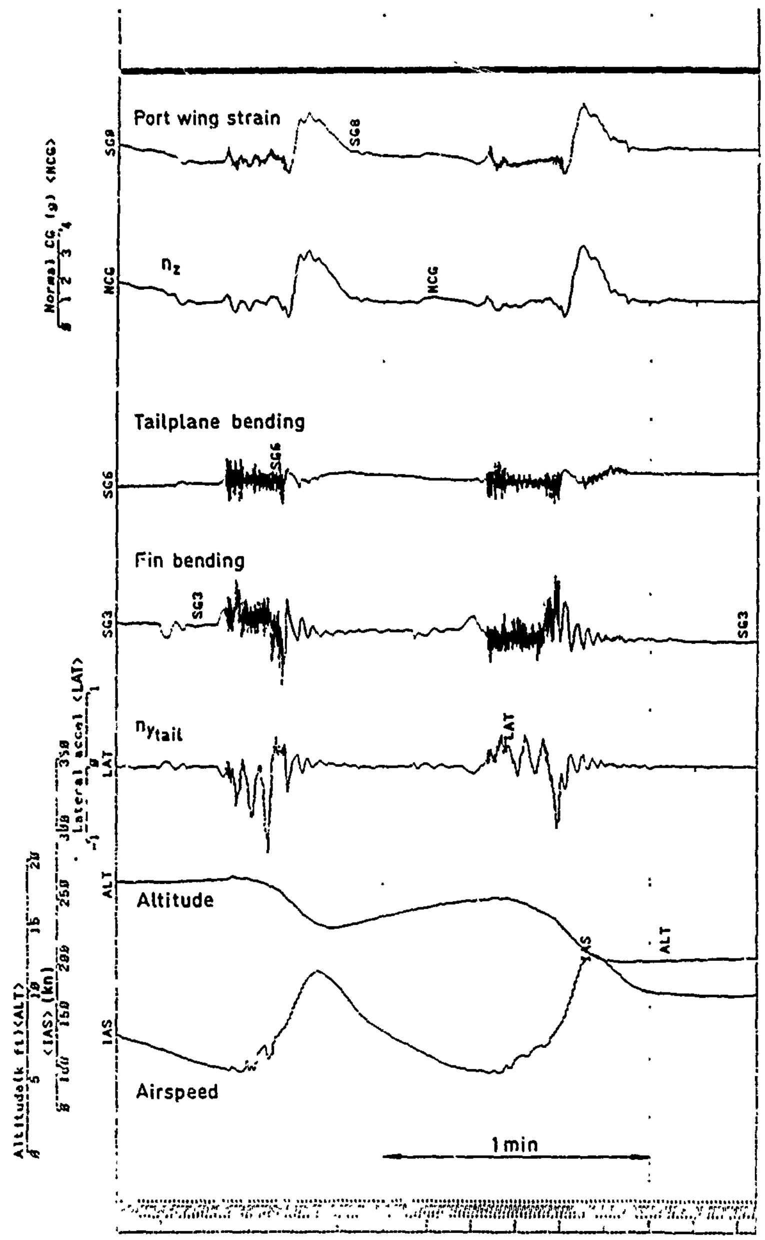


Fig 8 Jet Provost: structural activity during spins

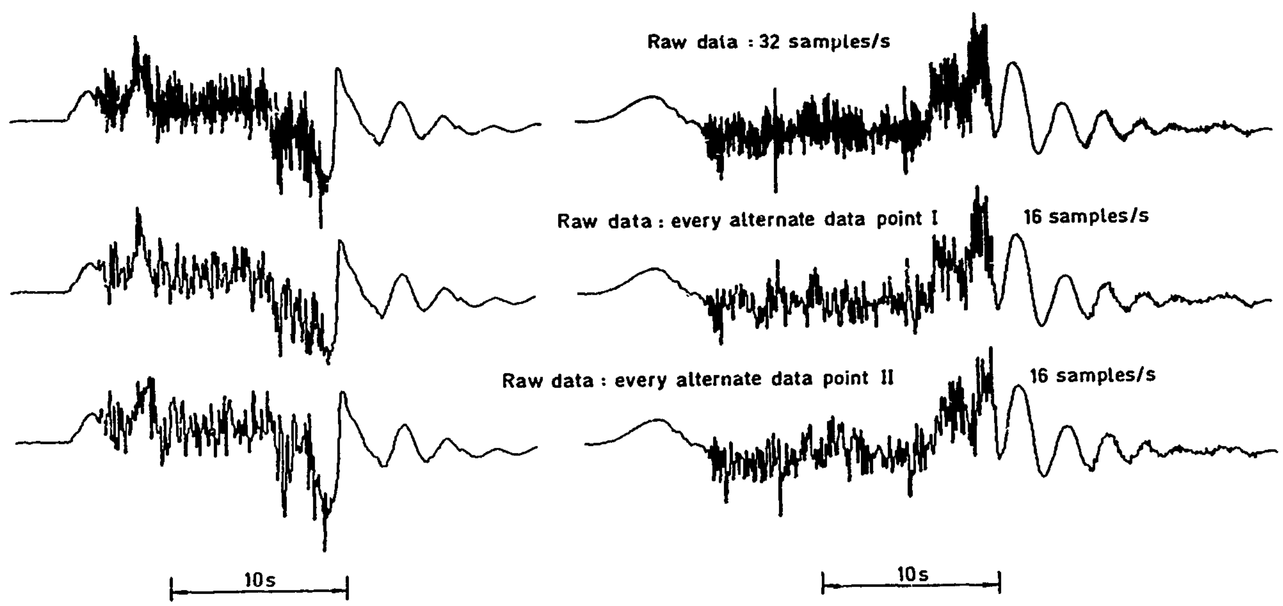


Fig 9 Jet Provost: fin bending moment during stalls and spins

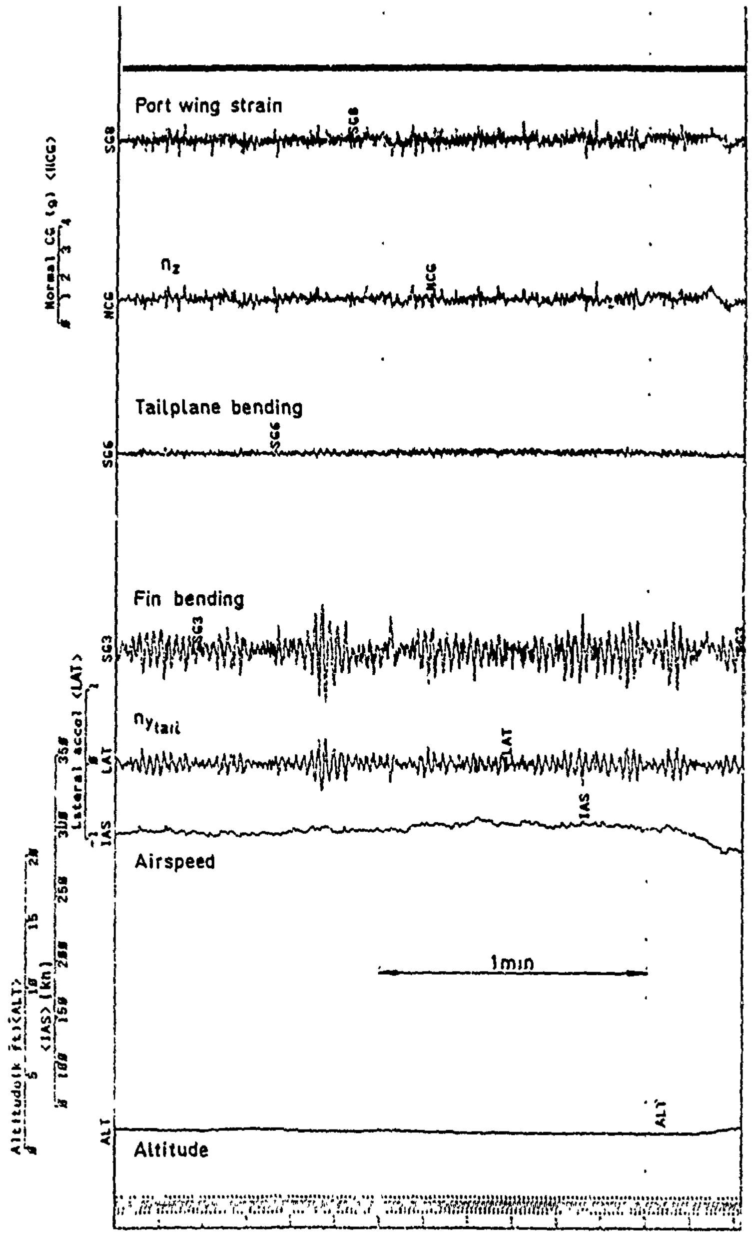


Fig 10 Jet Provost: typical record from a low level flight

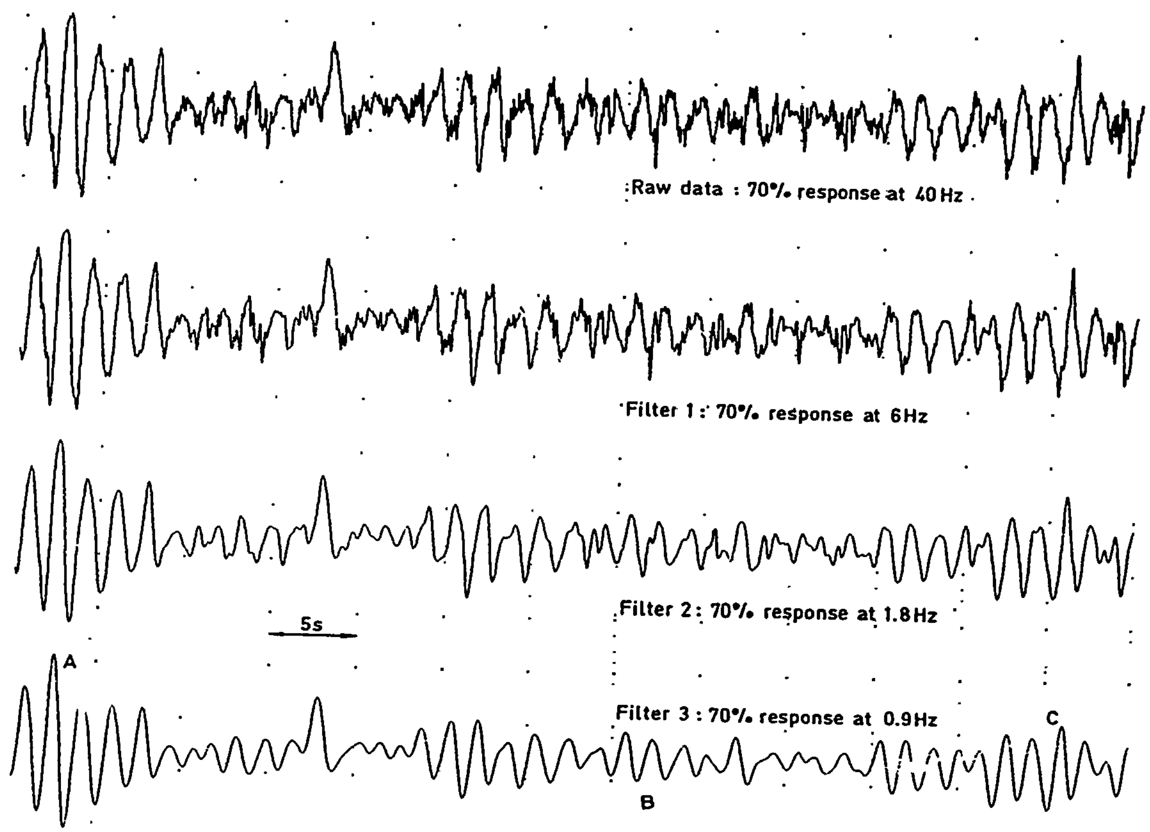


Fig 11 Jet Provost: effect of frequency bandwidth on output of a fin gauge during low-level flight

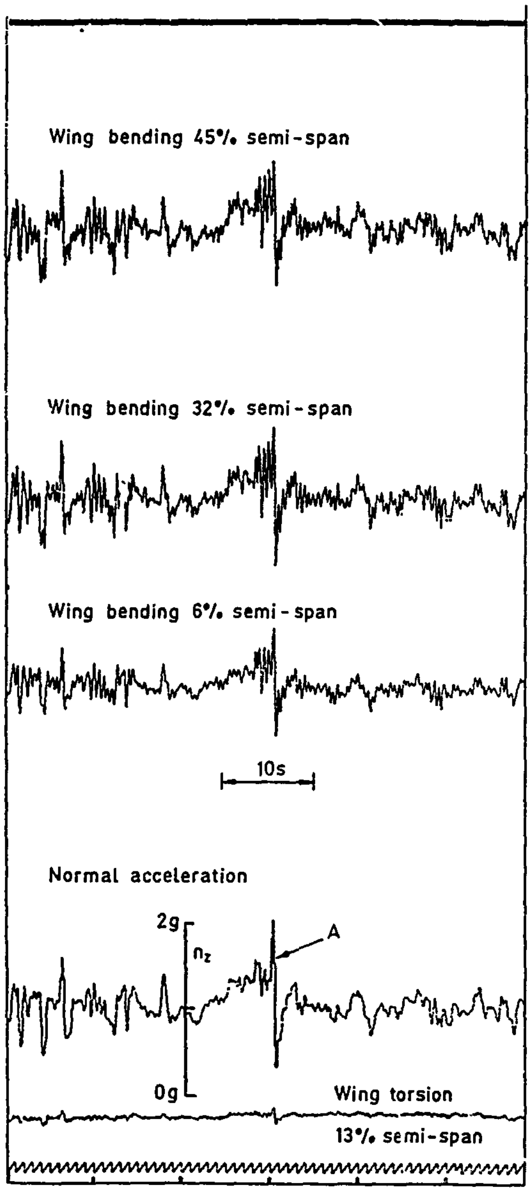


Fig 12 Strain gauge outputs and normal acceleration on Hercules: IAS ≏ 210 kn, altitude < 1500 ft, 3/8 cumulus at 1800 ft and mean wind 17 kn

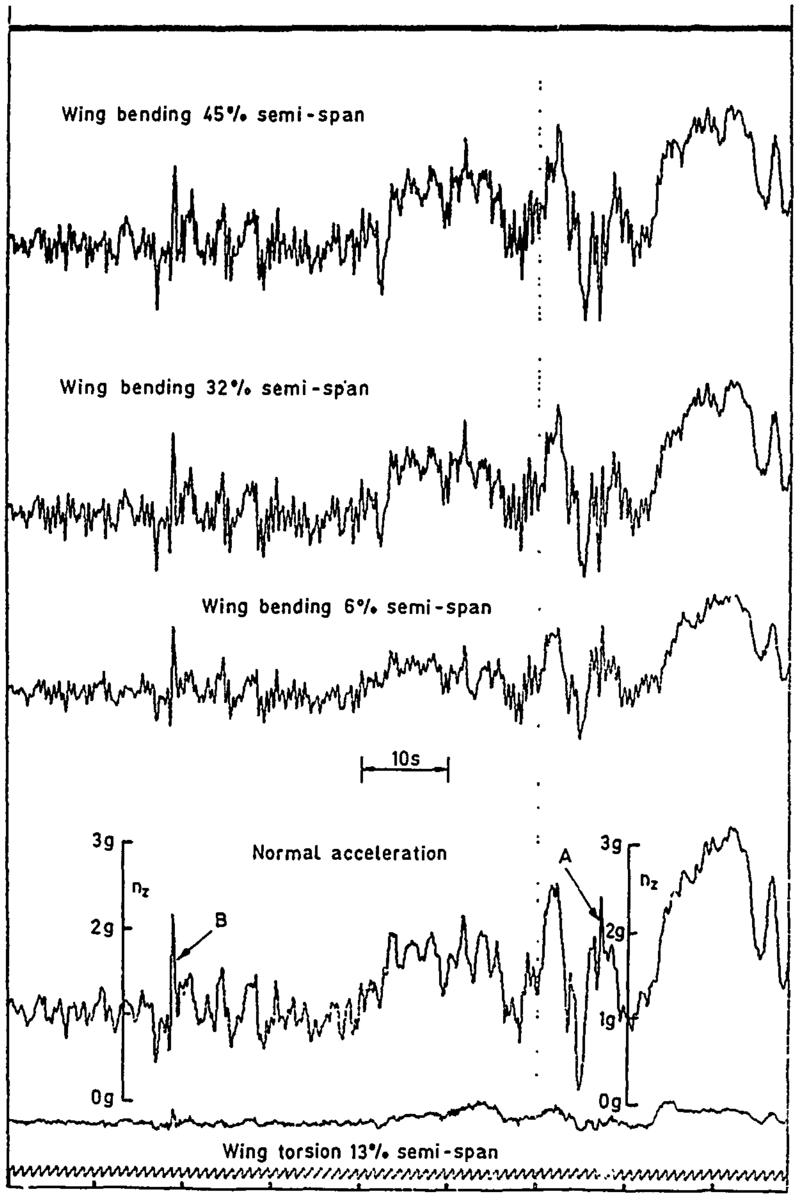


Fig 13 Strain gauge outputs and normal acceleration at cg measured on Hercules during a combat training exercise: IAS ≏ 210 kn, altitude < 1500 ft

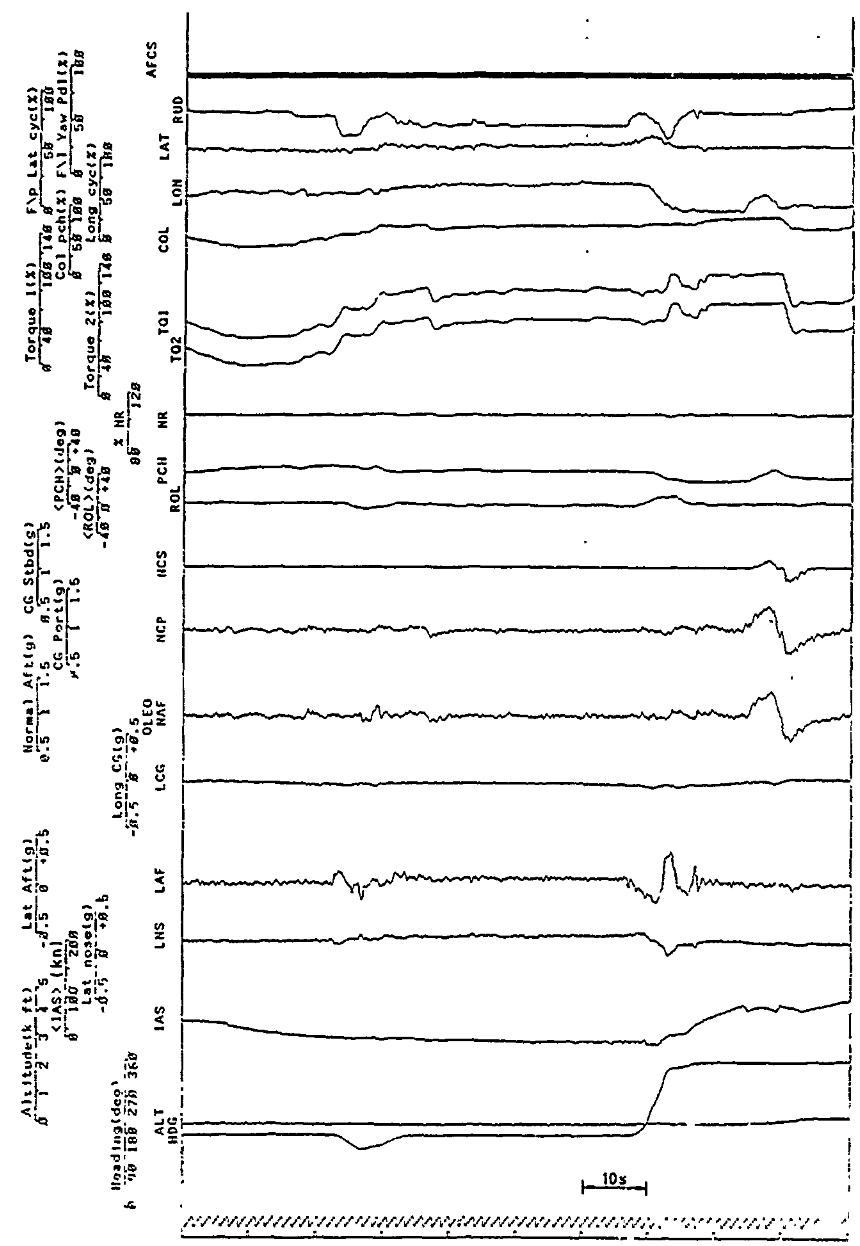


Fig 14 Sea King rapid turn from rearwards flight

THE F-15 FLIGHT LOADS TRACKING PROGRAM

J. T. Johnston
Branch Chief - Technology - Structural Dynamics and Loads

and

R. E. Pinckert/Branch Chief - Technology - Strength
R. A. Melliere/Unit Chief - Technology - Strength
McDonnell Aircraft Company
McDonnell Douglas Corporation
P.O. Box 516, St. Louis, Missouri 63166
USA

SUMMARY

The F-15 flight loads tracking program is a vital part of the USAF "Aircraft Structural Integrity Program (ASIP)."

The tracking program consists of four phases; data collection, data reduction, fatigue damage analysis, and fleet management. Data collection is accomplished with a multi-channel recorder and a load factor (g) exceedance counter. Twenty percent of the fleet is equipped with the multi-channel recorder and every F-15 has a g exceedance counter installed. After the fleet data have been reduced by the Air Force it is sent to McDonnell Aircraft for conducting fatigue damage analyses. Quarterly "Service Aircraft Fatigue Estimate (SAFE)" reports inform the Air Force how much fatigue life has been expended on each aircraft. These reports are used for establishing inspections and aid in fleet management of the F-15 Eagles.

This mature program, approaching 1,000,000 flight hours, has been valuable for solving in-service structural problems, understanding why the problems occurred, developing repairs, and for redesign. Future aircraft design will also benefit from this information.

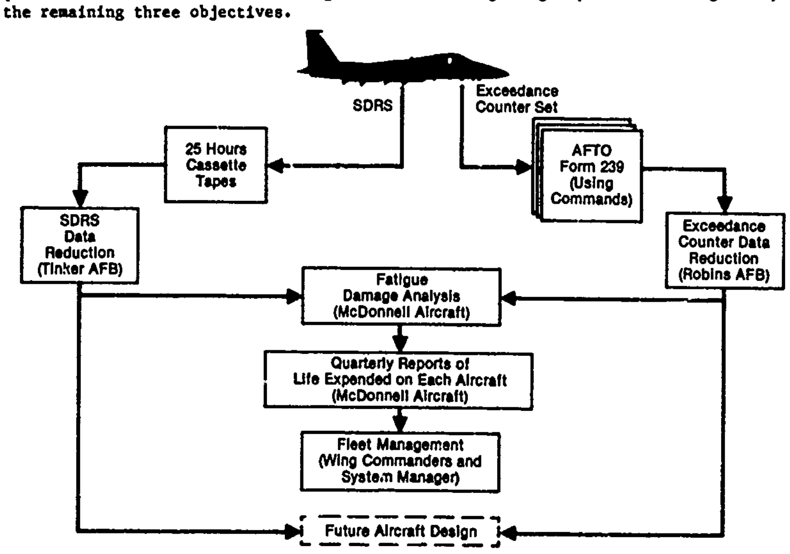
1. INTRODUCTION

The F-15 flight loads tracking program is an on-going process for recording flight parameters for operational aircraft, converting those parameters into "airframe load forces" applied to the aircraft during flight, and calculating the percentage of structural fatigue life expended. Through ASIP, the customer can evaluate aircraft mission utilization and maintenance scheduling requirements.

Specifically, the F-15 ASIP is a comprehensive plan centering around four important objectives:

- o To establish, evaluate, and substantiate airframe strength and durability (structural integrity).
- o To assess continuously the in-service integrity of individual airplanes by utilizing operational usage data.
- o To provide a basis to establish logistic support and aid in planning future aircraft utilization (maintenance, inspection, supplies, rotation of airplanes, and system phaseout).
- o To collect usage data to aid in development of improved structural criteria and methods of design, evaluation, and substantiation for future aircraft systems.

The first of the above objectives, structural integrity was attained during the design, test, and development phase of the F-15. The F-15 Flight Loads Tracking Program, shown in Figure 1, contributes directly to the remaining three objectives.

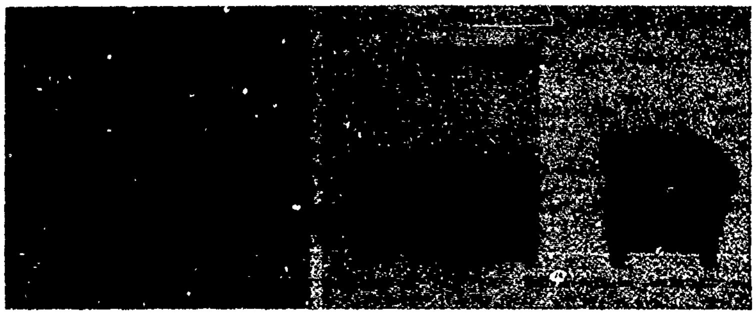


22CFR125.11(a) Applicable

Figure 1. F-15 Flight Loads Tracking Program

2. DATA COLLECTION

The data collection phase is one of the crucial parts of the flight loads tracking program for the remaining phases are only as reliable as the data collected. Since the establishment of service-life expectancy is dependent on the accuracy of the loads spectrum and actual aircraft utilization, the aircraft requires a multichannel recorder system. On the F-15, data is collected utilizing a multichannel Signal Data Recorder Set on 20 percent of the fleet and a load factor exceedance counter recorder on 100 percent of the aircraft. These two units are shown in Figure 2.



Tape Cassette (Left) Records Twenty-Two Flight Data Points Gathered by the Multichannel Signal Data Recorder Set/SDRS (Center). The Exceedance Counter Set (Right) Records Positive and Negative "g". GP43-0080-2

Figure 2. Signal Data Recorder Set and Exceedance Counter Recorder

In the initial design phase, MCAIR conducted trade studies to determine the optimum concept. The loads equations for all major structural components was the starting point. These equations defined all the flight parameters, control surface positions, angles, rates, accelerations, dynamic pressures, Mach numbers, etc. needed for design of the airframe. Then through regression analysis, data was deleted until the level of quality results dictated the number of data required. In addition, other forms of direct measurements were investigated, such as scratch gages and uncalibrated strain gages. These were deleted after consideration of the environmental conditions that exist where the gages would be placed and the reliability of such instrumentation. MCAIR continues to study the optimum method of obtaining flight information for generating loads information. To date the F-15 approach is still considered to be the most economical and reliable method.

The SDRS automatically records significant flight parameters and control surface positions. A total of 22 flight parameters, shown in Figure 3, are recorded continuously. Note that the sampling rates (the times a parameter is recorded per second) vary depending on the rate of change and importance of the parameter. With these data, loads time histories can be computed on any of the major structural components. Currently, loads are generated for the wings, stabilators, vertical tails, ailerons, and the forward fuselage. Other locations can be included with the change in the computer program. For example, there is currently a change in progress to add another fuselage station to the program.

The SDRS records the data on a cassette with a 25-hour recording capability. Documentary data (flight date, mission code, aircraft serial number, squadron number, and weapon identification) is manually entered by the pilot or ground crew before each flight. All other data is automatically recorded. When the magnetic tape in a cassette has been expended, the cassette is removed by the using command and sent to Tinker AFB for reduction.

The Exceedance Counter provides individual aircraft load factor data. It consists of an accelerometer transducer located near the aircraft's nominal center of gravity and a counter display unit. The transducer continuously measures the aircraft normal load factor while the counter display unit automatically records and displays the number of times the aircraft has been subjected to each of seven load factor levels; three negative (-2, -1, 0) and four positive values (3, 4, 5, 6, and 7..).

Exceedance Counter readings, together with flight log information, are recorded manually by the using command after each flight. The forms have been designed in such a way that they can be read automatically through the use of optical scanning equipment. Completed forms are sent to Robins AFB for data reduction.

The mission type and aircraft gross weight affect the amount of damage caused by a given load factor occurrence. Therefore, flight log information recorded on the forms includes data necessary to associate load factor occurrences with mission type and average aircraft gross weight. The combination of load factor occurrences, aircraft gross weight, and mission type for a given flight is converted to percent of fatigue crack growth life expended during the "fatigue damage analysis phase" of the flight loads tracking program.

As major aircraft components are changed they are recorded by the using command and the records forwarded to Robins AFB for data reduction. This provides the way to monitor the fatigue damage on individual serialized aircraft components which were removed for repair or overhaul and installed on a different aircraft or on the opposite side of the same aircraft.

Signal Source	Aircraft Parameter	Unit of Measure	Sampling Rate/sec
Central Computer	Date	—	—
	Mission Number	—	—
	Aircraft Serial Number	—	—
	Squadron	—	—
	Altitude	ft	1
	Velocity	kts	1
	Angle-of-Attack	deg	10
	Weapon Count	Item/Station	1
	Vertical Velocity	ft/sec	5
	Gunfire	Rounds	1
Exceedance Counter	Vertical Acceleration	g	30
Automatic Flight Control System	Stabilator Deflection - Right	deg	10
	Stabilator Deflection - Left	deg	10
	Control Augmentation System (3 Axis)	On-Off	1
Fuel Gaging System	Fuel Quantity	lb	1
Aircraft Sensors	Aileron Deflection - Right*	deg	30
	Aileron Deflection - Left*	deg	30
	Rudder Deflection*	deg	10
	Speedbrake	In-Out	1
	Wheels	Up-Down	1
Signal Data Recorder	Roll Rate	0/sec	30
	Pitch Rate	0/sec	15
	Yaw Rate	0/sec	10
	Roll Acceleration	rad/sec^2	30
	Lateral Acceleration	g	5
	Longitudinal Acceleration	g	5
	Time	sec	1
		Total	239

*Transducers recording these parameters were added to the aircraft specifically for the SDR

GP[illegible]

Figure 3. Flight Parameters Continuously Recorded on SDR Cassettes

3. DATA REDUCTION

Tinker AFB processes the SDRS cassette tapes using ground playback equipment and computer programs. The first step is to reformat the signal data into engineering units and transcribe onto a computer compatible magnetic tape, edit nonsignificant data such as non-maneuvering data, and identify erroneous data such as malfunctioning transducers. The compatible tape is then processed through an IBM computer using the F-15 Operational Flight Loads Computer Program (OFLCP) to generate the applied loads, solve for the internal loads, and generate stress spectra and summary tables of average usage. A flow diagram of this process is shown in Figure 4.

Exceedance Counter/flight log data and component tracking data is reduced by Robins AFB personnel. Optical scanning equipment is used to extract data from the forms. The Exceedance Counter/flight log data is then checked for validity and arranged into a flight-by-flight time sequence using a computer technique.

Periodically the SDRS data reduced by Tinker AFB, and the Exceedance Counter/flight log data and component tracking data reduced by Robins AFB, is sent to McDonnell Aircraft Company for fatigue damage analysis.

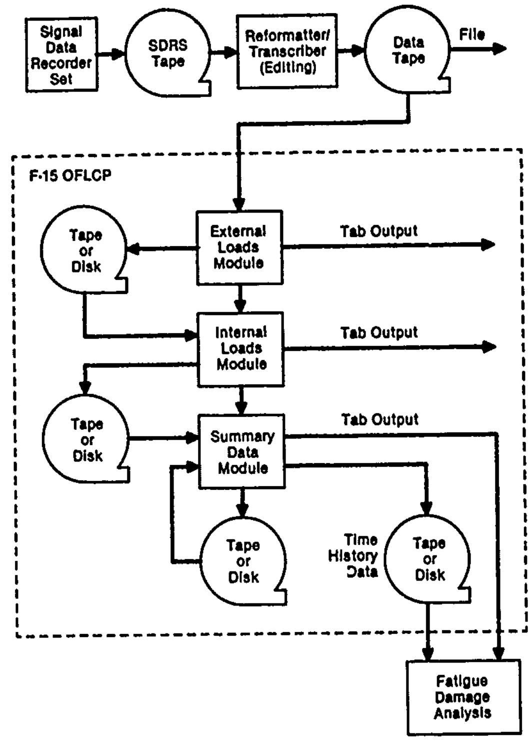


Figure 4. F-15 OFLCP Functional Flow Diagram

4. FATIGUE DAMAGE ANALYSIS

The F-15 fatigue damage analysis involves determining how much aircraft fatigue crack growth life has been expended by the wear and tear of day-to-day missions. Initial flaws are assumed to exist at critical locations throughout the aircraft. The flaw growth is tracked using analysis spectra developed from SDRS data and Exceedance Counter/flight log data together with the Contact Stress crack growth model of Dill and Saff (Reference 1). The Contact Stress model accounts for residual stresses caused by peak overloads and for crack growth retardation caused by crack tip plasticity and subsequent crack closure stresses. The fraction of crack growth life expended (the accumulated damage) is expressed as a crack growth damage index, where a damage index of 1 indicates that the assumed initial flaw is predicted to have propagated to its critical crack size in the structure. Damage estimates for 16 locations (Figure 5) on each aircraft are made using a computer program which determines and totalizes the damage for each maneuver. This is accomplished using stress spectra developed from Exceedance Counter and SDRS data.

For locations which have a primary relationship to vertical load factor, (Nz), e.g., inner wing locations spectra are developed from Exceedance Counter data available on a flight-by-flight basis from the flight log forms reduced at Robins AFB. Each recorded load factor is converted to a stress, based on the mission type and aircraft gross weight for the mission. The conversion from load factor to stress is based on stress/load factor relationship developed from SDRS data. A representative stress spectrum developed from Exceedance Counter data for each flight is shown in Figure 6. The highest load factor is coupled with the most negative, and the next highest with the next lowest, etc. After using up all negative occurrences, the remaining positive occurrences are coupled with a preset stress level representative of a 1.0 g occurrence. Similar stress spectra for each flight are then processed through the damage computer program to calculate incremental crack growth damage for the flight.

For locations which do not have a primary relationship to Nz, e.g., aileron backup structure crack growth damage is determined using flight log and Exceedance Counter data supplemented by average damage rates obtained using SDRS data.

Damage for individual removable aircraft components is tracked with the aid of component tracking information reduced by Robins AFB. Damage accumulation estimates are updated and reported quarterly to the Air Force by McDonnell Aircraft Company for each operational F-15 aircraft.

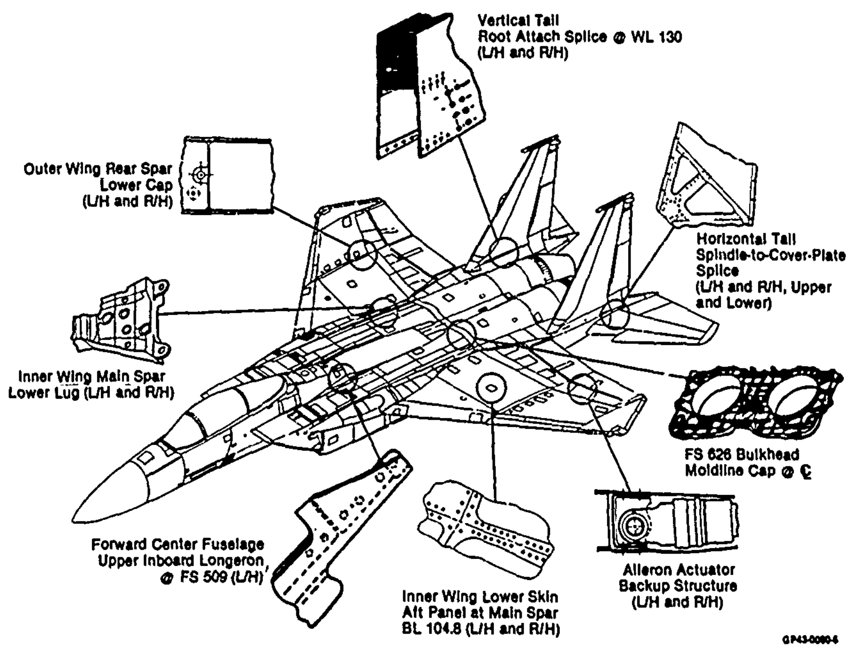


Figure 5. Fatigue Critical Locations Tracked Directly Using SDRS and Exceedance Counter Data

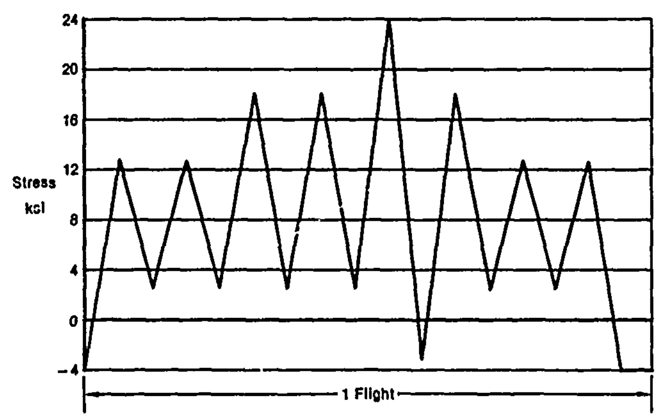


Figure 6. Typical Flight Profile Developed From Exceedance Counter Data

5. SAFE REPORTS

Quarterly Service Aircraft Fatigue Estimate (SAFE) reports are processed by McDonnell Aircraft Company and submitted to the USAF to inform them as to how much crack growth damage has accumulated on each aircraft in the fleet. The SAFE reports present summary tables listing the damage indices and inspection projections. Crack growth damage indices are calculated for 44 locations on each aircraft. The 44 locations include the 16 direct locations identified in Figure 5, for which spectra are available from SDRS and Exceedance Counter Data, plus 28 adjacent locations. Crack growth damage indices for the 28 adjacent locations are obtained from the 16 direct locations through the use of transfer functions which relate damage at the adjacent locations to damage at the direct locations. Figure 7 presents en example listing of a SAFE report damage index table for the 16 direct locations. Damage indices are listed for the current quarter (previous 3 months) as well as cumulative to date. Inspection projections for each aircraft are made on the basis of the individual damage index for each location and an assumed damage accumulation rate. For projecting inspections at a given tracked location on a given aircraft, the average damage accumulation rate for that location and average flight rate for the squadron are assumed. Inspection projections for each aircraft and location are listed in the SAFE reports as shown in Figure 8.

Fleet Summary of Damage Indices for Direct Locations
Last Reporting Date - June 1983

Aircraft Serial Number	Total Hours	Aircraft Side	Inner Wing Lug	Inner Wing Skin	Outer Wing Rear Spar	Aileron Backup Structure	Fwd Center Fuselage Longeron	FS 626 Bulkhead	Stabilator Spindle Upper	Stabilator Spindle Lower	Vertical Tail Root	Damage Accumulation Period
760190	1,584	L/H	0 3486	0.1794	0 0912	0 3175	0 1081	0 1984	0 0759	0.0705	0.1700	Cum to
		R/H	0.3576	0 1860	0.0875	0 3246	—	—	0 0809	0 0743	0.1753	Date
		L/H	0.0165	0 0087	0.0053	0 0150	0 0052	0 0094	0.0033	0 0025	0.0073	Current
		R/H	0 0169	0 0092	0 0054	0.0153	—	—	0 0037	0.0033	0 0078	Quarter

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Figure 7. Safe Report Example - Damage Index Summaries

Inspections Required Within Next 10 Years
Last Reporting Date - June 1983

Aircraft Serial Number	Location	Cum Damage Index	Aircraft Total Hours	Next Projected Inspection Date		
				Aircraft Hours	Quarter	Year
760190	Inner Wing Lug L/H	0 3476	1,584	4,532	1	1993
	Inner Wing Lug R/H	0 3486		4,491	4	1992

GP43-0080-8

Figure 8. Safe Report Example - Inspection Projections

6. SOLUTION OF IN-SERVICE PROBLEMS

The SDRS data is also used frequently to assist in solving F-15 structural problems that occur in-service. Based on SDRS data, engineering analyses can be performed to determine why cracking occurred in-service, and to determine how a retrofit or design change should be implemented to prevent similar problems in the future.

The following examples of how SDRS data was used to solve in-service problems show how very important the information is in assessing fleet usage and in performing fatigue analyses of the F-15:

- Upper-Inner Wing Skin Buckles - SDRS analyses of wing bending and torsion provided information to determine the cause of upper-inner wing skin buckling. As a result, skins were beefed-up on future models, and a caution notice was published warning the pilots about excessive overloading of the airplanes. An aural Overload Warning System has since been developed to warn the pilot of an approaching overload condition.

- Vertical Tail Buffet - SDRS data was used to define the angle-of-attack where unsteady flow from the wing impinges upon the vertical tail and results in fin buffet (Figure 9). After performing loads and fatigue analyses and conducting tests, changes in the upper vertical tail design were made to reduce the probability of structural cracking.

- Upper-Outer Wing Buffet - Analyses of upper-outer wing skin cracking were performed using SDRS data as a basis. The primary cause of the cracking was the environment resulting from local flow separation. Design changes based on fatigue analyses and tests have since been made to the upper-outer wing skin and some wing ribs.

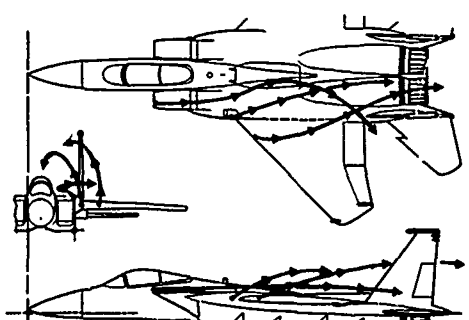

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Figure 9. Vertical Tail Buffet

7. FLEET MANAGEMENT

One of the objectives of the F-15 flight loads tracking program is to aid in fleet management. This is accomplished in two ways:

- Usage data provides the means to assess effects on remaining fatigue life when a new mission is dictated for all or part of the fleet.

- Usage data provides the means to assess effects on remaining fatigue life when a new mission is dictated for all or part of the fleet.

- The quarterly SAFE reports identify to the Wing Commander and the System Manager (Robins AFB) by tail number how much crack growth life has been consumed by prior flight history. The Wing Commander, at his discretion, can schedule his aircraft missions to even out crack growth life consumption. This may involve reassignment of an aircraft to missions where the usage is less strenuous. The Wing Commander will also be able to schedule major compliance more effectively. Similarly, the System Manager can use the damage indices to schedule aircraft for Analytical Condition Inspections (ACI). The SAFE report inspection projections are used to tailor the fleetwise fatigue inspections for critical areas to the usage of each individual aircraft. The inspection manuals are updated quarterly to list those aircraft and those locations that are due for inspection during the next calendar year.

8. FUTURE DESIGN

The flight measured loads data collected in the F-15 and in other fighters flight loads tracking programs will be of considerable help in establishing requirements for future fighter designs. In 1969 when the F-15 requirements were established, we had only limited load factor data. Now, we have an understanding of the relationship between pitch-roll-yaw at speeds, altitudes, and gross weights for different missions. The next USAF fighter will benefit from this information.

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STRUCTURAL LOAD MEASUREMENTS ON A NORTHROP NF-5A

by
D.J. Spiekhout
National Aerospace Laboratory NLR
Anthony Fokkerweg 2, 1059 CM Amsterdam
The Netherlands

SUMMARY

Load measurements were carried out on an NF-5A aircraft instrumented with a large number of strain gage bridges in the wing and the tail surfaces.
Measurements included:
* Specific stationary manoeuvres
* Specific dynamic manoeuvre conditions
* Complete mission segments typical for RNLAF-operational conditions.

Using ground calibration results, recorded strains were converted to sectional loads and with flight parameters like V, H, acc. etc. stored in a Structural Load Data Base (15 flights covering 7 configurations and 715 measuring runs). By means of an interactive computer program the data stored in this data base are easily accessible. The results can be presented in tables or in graphical form.
The Data Base may be used to evaluate the effect of changes in operational procedures, stores and store configurations on the fatigue load experience in various structural areas.

The paper gives a general description of the instrumentation used, the data handling and the flight conditions that were recorded.

The possible use of the data base is illustrated by means of a number of examples.

1. INTRODUCTION

In the design of modern fighters fatigue is an important item. Based on a number of design assumptions with respect to usage and loading environment a crack free service life has to be determined. More recently, the damage tolerance concept has been introduced, in which inspection intervals have to be determined in such away as to find a growing crack before it becomes critical. During the life of a fighter the usage and loading environment may change because of changes in role or tactics. This has also been experienced with the NF-5 aircraft and for this reason a load monitoring system using counting accelerometers has been active since several years. One of the disadvantages of the counting accelerometer is that no information is collected about the sequence of the load peaks. Furthermore, the actual strain in a fatigue critical part is dependent on load factor and other quantities such as total mass and mass distribution.
Main goal of the presently described flight program was to get more information about wing loads and the load distribution over the wing.
Of special interest is the correlation between load factor and strain at critical locations. Sectional loads have been calculated from recorded strains for four wing sections.

In addition, strain histories in some fatigue critical locations were recorded.
Besides loads of the wing, vertical and horizontal stabilizer loads have been measured in the program.
In the flight program a large number of measuring runs have been recorded during 15 flights. Besides the normally used mission segments in the operational task of the NF-5 a large number of specially prescribed manoeuvres have been flown. After data processing and convertion to engineering units the information has been stored in a data base, which is available for further analysis. In chapter 2 the instrumentation will be presented, whereas in chapter 3 the data processing will be discussed.
The flight program and application of the data base will be presented in chapter 4 and 5 respectively.

2. INSTRUMENTATION

In order to be able to do various flight trials, including e.g. certification of new external stores, one NF-5A of the RNLAF has been equiped as a test-aircraft, figure 1. This equipment includes an extensive instrumentation package. In addition, an "instrumented wing" is available, which was mounted to the test aircraft for the presently reported measurements.

During manufacturing of the wing at Canadair a large number of strain gage bridges have been installed. Further an instrumented boattail with strain gage bridges in the horizontal stabilizer was available and has replaced the original one.
For measuring loads on the vertical stabilizer strain gage bridges have been installed by NLR.
The total instrumentation package used for the present measurements can be divided into the following groups:
- Basic instrumentation
- Strain gage bridges at wing, horizontal- and vertical stabilizer
- Instrumentation to check the quality of the recordings

Full listing of the recorded parameters is presented in table 1.

2.1 Basic instrumentation package

From the available instrumentation in the NF-5A test aircraft a selection has been made in order to measure the aircraft movement, attitude and configuration of the aerodynamic controls. The movement and attitude of the aircraft is defined by speed, altitude, three linear accelerations and three attitude angles. The aerodyamic configuration of the aircraft is defined by the position of the control surfaces.

2.2 Strain gage bridge instrumentation

The so called "instrumented wing" has been supplied with a large number of strain gages and associated wiring during manufacturing of the wing at Candair Montreal. The strain gages were situated in four wing sections, namely 32 LH, 32 RH, 68 RH and 104 RH, see figure 2. The strain gages were manufactured by Baldwin. By internal wiring full bridges with four active strain gages were made. In most cases the bridges are located two by two giving prime and spare bridges. Further the configuration of the bridges is such that some bridges are especially sensitive to vertical shear force, others to bending moment or torsion moment in the wing, see figure 3. In addition two half bridges have been installed by NLR at wing section 11 LH. The location of one of these bridges has also been used in strain measurements during actual operational flight with an NF-5. These bridges were manufactured by TML. In total 63 bridges were available in the wing. In the present program the output of 40 strain gage bridges was recorded, respectively 2 at WS 11, 10 for both inboard wing stations and 9 for both outboard stations. In the instrumented boattail a total number of 22 full bridges was available. For the present program it was decided to use one bending and one shear force bridge in both the LH and RH horizontal stabilizer. The purpose of using the output of these bridges was to get an impression of the loading environment of the horizontal stabilizer during the different measurements.
In this case again Baldwin gages have been used.
The last group of strain gage bridges are located at the vertical stabilizer. These bridges have been installed by NLR using strain gages manufactured by TML. Besides one bending- and one torsion bridge, which were full bridges, also three half bridges have been used near the fin skin fillet radius. This is one of the points of concern in the structure of the NF-5 with respect to fatigue.

2.3 Instrumentation to check the quality of the recordings

For this purpose four channels have been used. Besides three channels for monitoring the supply voltages (10.2 V) of the strain gage bridges in wing and stabilizers on one channel a trapezium shaped signal, which is generated continuously was recorded.
Further, two temperature sensors were installed at the lower and upper skin of the LH wing at wing station 75.

2.4 Signal conditioning and recording instrumentation

Signal conditioning consisted of signal filtering and signal amplification. For the strain gage bridges analog 20 Hz filters were used with a decrease in sensitivity of 12 dB/octave for higher frequencies. For the acceleration signals 5 Hz filters with a decrease in sensitivity of 24 dB/octave have been used.
Data were recorded by a PCM (Pulse Code Modulation) recorder on a 16 track magnetic tape with a width of one inch. The PCM recorder has been manufactured by Radix Telemetry Corporation and has the capability of recording up to 72 channels with a sample rate of 138 times per second per channel. The word length of the system is 10 bits.

3. DATA PROCESSING AND GROUND CALIBRATION

In this chapter the data flow will be described shortly from recording on the PCM recorder until storage of the data in a data base. Data reduction is very important in order to reduce computer time and permanent file costs in the subsequent data processing and analysis of the measurements in the future.

3.1 Data processing

In figure 4 the data flow is presented.
In the program no continuous recording is performed. Before each measuring run the pilot had to switch the PCM recorder on. Without any data reduction the total amount of data should be very hard to handle and the computer time needed will be high. Therefore data reduction was very important. The different steps in the data flow will be explained shortly.
After each flight the PCM tape was removed from the aircraft and traces were made by means of a quick look facility of the following parameters: vertical acceleration, speed, altitude, flaps/event marker, PCM test signal and the 3 monitor bridges for the power supply of the strain gage bridges. This quick look plot was used by pilot and project engineer for checking the quality of the recordings and in order to judge if the objectives at the different runs has been met. If so the PCM tape was sent to NLR.
As a first step the PCM tape was converted to computer compatible tape for the NLR Cyber computer. At the same time, time histories were made of the parameters: vertical acceleration, altitude, speed, flaps/even marker, temperature lower skin, one bending bridge of wing station 11, horizontal stabilizer LH and vertical stabilizer. The plots included more parameters and were of much better quality than the Quick Look plots mentioned earlier. Along the X-axis a time scale was presented using framecount numbers. Reference lines on the plot made measurement of the magnitude of the various signals possible. The project engineer selection from this plot the relevant parts of each run for further data processing. From the original recorded data of 229 minutes a reduction to 91.5 minutes was reached. At the same time the sample frequency of the PCM recorder was reduced with a factor of four to 35 times per second. This sample rate was considered sufficient as structural responses in modes with frequencies above 5 Hz were expected to be negligible. The next step in the data processing was performed by a computer program called "LOADMES". This program performed a convertion to engineering units for all parameters. Also for each sample the total mass of the aircraft was calculated. This parameter was not recorded, therefore the pilot made recordings by hand of the fuel consumption at the end of each measuring run.
Also the take off mass of the aircraft was known. By linear interpolation between the "measuring points" the instantaneous mass of the aircraft has been calculated. Further the computer program calculated the sectional quantities as moments and forces for the wing and stabilizers using the output of the strain gage bridges. The relations between strains and sectional loads have been determined in ground calibration

tests which will be discussed in chapter 3.2. More details of these calibrations are presented later.
A third data reduction was achieved by decreasing the number of parameters to 38. It was decided to store in the data base for the four main wing stations only the derived shear force, bending moment and torsion moment, rather than the output of all the individual strain gage bridges. For the vertical- and horizontal stabilizers the same approach has been taken. In total only three strains are stored: namely two at wing station 11 and one near the critical radius of the vertical stabilizer.
Also the monitoring bridge signals and the PCM test signal were deleted from the data base as they only served to ensure data quality in earlier steps. In table 2 the resulting list of parameters which are stored in the data base for each sample is presented. The total amount of data actually stored is only 5 percent of originally recorded data.
It should be mentioned here that the original PCM tapes are kept in the archives of NLR.
If necessary the information can be made available for research in the future.

3.2 Ground calibration

For calibration of the strain gage bridges in wing and stabilizers ground tests were performed. By actually loading the aircraft structure relationships between the output of the strain gage bridges and sectional quantities as shear force, bending- and torsion moments could be derived.

For the wing an extensive ground calibration has been performed at the NLR facility in the North East Polder. The wing box structure was loaded by pulling down the center section of the wing by a hydraulic cilinder on two columns with pads, which were placed symmetrically at left and right hand side under the wing. Symmetrical loading over both columns was ensured by a substructure connected to the four attachment points of the wing, which allowed free "rolling" of the wing, see figure 5. Loads have been applied at 48 loading points using combinations of roughly 6 wing stations and 7 chord wise positions, see figure 6. After installation of the wing on the aircraft an additional limited ground calibration was performed using less loading points.
The measured relations between strain bridge outputs and known applied sectional loads were used to derive "best fit" equations to calculate sectional loads from strains. As an example, table 3 gives the derived coefficients for one wing station.

For the vertical stabilizer a ground calibration was performed using a substructure which loaded the central spare area with a compression force and at the same time the upper rudder hinge with a tension force. In this way reaction forces on the calibration rig were kept small. Both forces could be applied simultaneously by one compression force on the substructure. In total 9 loading points have been used. Three of them introduced pure bending, four pure torsion and two combinations of bending and torsion at loading points similar to Northrop design conditions. As a result coefficients for calculating the sectional quantities from the recording strains were found. It was found that for smaller bending moments a large part of the bending moment was transferred by the attach angles and formers into the fuselage. The same had been observed in similar measurements carried out by the CAF in Canada.

For the horizontal stabilizer the ground calibration was performed by loading the stabilizer by means of a pad. Loads have been applied at 5 loading points at left- and right hand stabilizer using 3 stabilizer stations and 3 chord wise positions.
Again coefficients for calculating the sectional quantities were derived.

4. FLIGHT PROGRAM

The main purpose of the present program was to get more knowledge about wing loads and load distributions over the wing of the NF-5 during operational RNLAF usage of the aircraft.
The load and load distribution is dependent of configuration and type of manoeuvre.
So the flight program had to cover the configurations and manoeuvres flown in the annual training program of the RNLAF. In this chapter the selection of configurations and manoeuvres will be described in more detail.

4.1 Selection of configurations

In general a store influences the loading of the wing in the following ways:
the additional mass of the store increases the total mass of the aircraft which has to be compensated by a larger total lift force on the wing. Besides this overall effect the store loads the attachment point with a point load due to its mass. These loads in combination with the aerodynamic drag forces may deform the wing and change the lift distribution over the wing.
From a life monitoring program using counting accerlometers, the normally used configurations were known.
In all configurations tiptanks were used.
Three configurations covered 96 % of all flights. These configurations had two 150 gallon tanks at inboard and at the center line pylon: none, one 150 gallon tank or a rocket bomb dispenser. Further an increased usage of the configuration with only one center line store was noticed. In order to make it possible to compare results with Northrop flight tests with the NF-5A aircraft, the configurations with only 5 pylons and the one with two 275 gallon fuel tanks at inboard pylons had to be taken into account.
In addition two more "extreme" configurations were used, namely the light clean configuration, no pylons, and a very heavy one with stores at the outboard pylons.
The clear configuration may be used as a reference condition for presenting the influence of stores. Stores with other mass configurations may be simulated by chaning the fuel quantity in the tanks.
In table 4 the whole "range" of configurations is presented.

4.2 Selection of manoeuvres/flight segments

For the present program 15 flights were available. Taking into account the available flight time for

each configuration and the "configuration mix" an overview of measurements to be flown has been made, see table 5. A number of groups can be distinguished, namely:
* 1 (1-4) A few measurements which were made for each flight. From the "zero g" measurements the reference conditions for the strain gage bridge will be derived. The ground runs and the combination of zero and one of runs allow checking of a proper function of the strain gage bridges. The in flight runs were made at 10000 ft with a velocity of 400 KIAS.

* 2 (5-20) A number of load factor values were measured during symmetrical pull up/push over manoeuvres and turns. Of course the maximum and minimum permitted load factors are dependent of the configuration. As can be seen in table 5 these measurements were made at the begin and the end of a flight in order to study the influence of fuel consumption on the load distribution over the wing. Further effects of configuration, flap position and load factor on the lift distribution over the wing in spanwise and chordwise direction can be studied. In general all these measurements have been flown at 10000 ft with a speed of 400 KIAs. In a few cases different altitudes and speeds have been used.

* 3 (21-49) Short and long inputs of aerodynamic controls. In the case of a short input the stick or pedal force is relieved at the moment that the maximum deflection has been reached. With these measurements effectiveness of aerodynamic control and the resulting loading of the aircraft as a function of configuration and mass variation can be studied. The same altitude and speed were used.

* 4 (50-65) Measurements were made during a number of flight segments which are commonly used, such as taxi, take off, landing etc for the different configurations. Also a few more unusual measurements were made, such as Mach run, stalls, inverted flight. Further side slips and rudder rolls were flown. These manoeuvres espcially load the vertical stabilizer.

* 5 (66-81)The last group of measurements consist of relevant RNLAF flight segments such as simulated attacks, range passes, basic fighter manoeuvres and air combat engagements. For those measurements only the normally used configurations were taken.

The whole set-up of the program is such that the measurements in group 2 and 3 can be used for analysing those in groups 4 and 5. The whole program consisted of 715 measuring runs, that were distributed over 15 measuring flights. In table 6 an example is given of a detailed program for one flight.

5. USE OF THE DATA BASE

As mentioned in previous chapters all flight data have been stored in a data base after data reduction and conversion to engineering units on a "per run" basis. In the data base the data pertaining to a run is preceded by a so called run information block which includes data like flight- and run number, date of flight and configuration.
The most simple form of data presentation is a direct printout of the data of one sample, see table 2. Time differences between succeeding samples can be easily changed. The smallest time differences is about 0.03 sec. Besides this printout various types of plots can be made using an interactive computer program. Besides time histories also "cross" plots of one parameter against another can be made.
Of course, the scale of a plot can be easily adjusted, and windowing of a plot is possible. Some examples are given in the figures 7-9. In figure 7 and 8 a few time histories are shown.
In figure 9 a cross plot of load factor against bending moment is presented. Interesting is the large variation in the "bending moment per g" relation, especially in the range from 1.5 till 3.5 g.
A plot like presented in figure 10 can be made very easily with the interactive computer program.
Figure 10 shows that the strain/g relation is dependent on the airspeed.

Since it became available, the data base has been used in a number of projects:

* In the first project an estimation has been made of the fin bending moment spectrum of the NF-5 for RNLAF usage.
As may be clear from chapter 4, measurements have been performed during typical RNLAF conditions. An example is given in figure 11. As a next step a peak between means counting was performed on the bending moment sequence for all relevant runs.
This resulted in spectra per mission type and by upscaling the resuls to the actual mission mix, which was known from a life monitoring program, a RNLAF fin bending moment spectra was "composed". By comparison with the real spectrum of the F-5E full scale fatigue test an estimation of the crackfree service life could be made.

* In a second project a study has been made of the influence of higher speed on the loads during a "range" mission. In figure 12 results are shown for a dive bombing attack using different initial speeds. Besides higher vertical accelerations which were experienced, also the influence of speed on the bending moment per g relation is shown.

* At the moment the data base is used for determining high loading conditions in the RNLAF operational usage.
From Northrop load envelopes for bending and torsion moment in the wing are known. All runs in the data base have been searched for exceedings of 60 % of the limit load envelope for the four wing stations. An example is shown in figure 13 of a part ACT flying and the loads in wing station 32. It is interesting to note the variation in the bending/torsion moment relation, as a result of speed variation.

TABLE 1

Review of recorded parameters

parameter	range	recording tolerance	Remarks
airspeed (indicated)	50 - 650 kts	± 5 kts	see chapter 2.1
altitude (fine and course)	0 - 32000 ft	± 30 ft	
vertical acceleration	-10 - +10 g	± 0.05 g	
longitudinal acceleration	-1 - +1 g	± 0.05 g	
lateral acceleration	-1 +1 g	± 0.05 g	
angle of pitch	0 - 360 degrees	± 2 degrees	
angle of roll	0 - 360 degrees	± 2 degrees	
grid heading	0 - 360 degrees	± 2 degrees	
position elevator	± max range	± 0.1 max range	
rudder pedal position	± max range	± 0.1 max range	
lateral stick position	± max range	± 0.1 max range	
speed brake selection	in -max range	-	
flap selection	up, trans., manoeuvre,full	-	
run number	0 - 99	-	
frame count (internal clock)	-	138 times per second	
event marker	-	-	
strain gage bridges in wing	-	-	see chapter 2.2
strain gage bridges at horizontal stabilizer	-	-	
strain gage bridges at vertical stabilizer	-	-	
temperature at wing (upper and lower skin)	-50 - +120 degrees Celsius	± 5 degrees Celsius	see chapter 2.3
monitor bridges (for supply voltage)	-	-	
PCM test signal	-	-	

TABLE 2

Parameters stored in data base for each sample (data frame)

par.nr.	parameter	unit
1	Frame count (internal clock)	ca. 1/138 s
2	Altitude	feet
3	Indicated airspeed	kts
4	Aircraft weight	lbs
5	Flap selection	switch pos.
6	Elevator position	degrees
7	Lateral stick position	-
8	Rudder pedal position	-
9	Speedbrake selection	switch pos.
10	Angle of pitch	degrees
11	Angle of roll	-
12	Grid heading	-
13	Vertical acceleration	g (x0.01)
14	Longitudinal acceleration	- -
15	Lateral acceleration	- -
16	Temperature at wing (upper skin)	degrees Celcius
17	Temperature at wing (lower skin)	" "
18-19	Strain gage bridge at WS 11, 18 %, 40 % chordwise	microstrain
20-23	Shear force at WS 32 LH, RH, 68 RH, 104 RH	N
24-27	Bending moment at WS 32 LH, RH, 68 RH, 104 RH	Nm
28-31	Torsion moment at WS 32 LH, RH, 68 RH, 104 RH	Nm
32-33	Shear force at root horizontal stabilizer LH, RH	N
34-35	Bending moment at root horizontal stabilizer LH, RH	Nm
36	Bending moment at root vertical stabilizer	Nm
37	Torsion moment in vertical stabilizer	Nm
38	Strain gage bridge at vertical stabilizer	microstrain

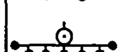
TABLE 3

The coefficients for the determination of the load components at wing station 104 RH (NF-5 instrumented wing)

% wing chord	18	21	24	27	40	42	66
Bridge type	Bending	Shear	Bending	Shear	Torsion	Bending	Shear
PCM channel	49	50	51	52	53	54	56
Coefficient for Shear Force	-	-53.5	3.8	-	13.5	-	-11.9
Coefficient for Bending Moment	-1.0	-	-	-1.6	4.4	11.4	-
Coefficient for Torsion Moment	-3.6	-3.0	-	-	9.2	-	5.7

TABLE 4

Configurations in flight program

config. code	Configuration 1)		number of flights	Operational weight empty 2) (lbs)	Average take-off weight (lbs)	Average flight durations (min)
1		clean	1	10356	14802	55
2		5 pylons	4	10987	15421	49 3)
3		one 150 US gallon tank	2	11134	16632	53
6		two 150 US gallon tanks	3	11289	17688	55
4		three 150 US gallon tanks	3	11436	18870	60
7		two 275 US gallon tanks	1	11361	18457	140 3)
5		three 150 US gallon tanks + two N-containers	1	12869	20284	105 3)
all			15			63

Note: 1) all configurations have tiptanks
2) 1 lb = 4.448 N
3) including one stop for refueling

TABLE 5

Overview of measurements in flight program

nr.	measurement \ configuration code / configuration	1	2	3	4	5	6	7
1	groundrun, full/empty tiptanks, before flight	+	+	+	+	+	+	+
2	" " after "	+	+	+	+	+	+	+
3	cruise, 0 and 1 g begin flight	+	+	+	+	+	+	+
4	" " end "	+	+	+	+	+	+	+
5	g values, symmetrical, flaps up, begin flight	+	+	+	+	+	+	+
6	" " " down, " "		+	+	+		+	+
7	" " flaps up, end flight			+	+		+	+
8	" " " down, " "				+		+	+
9	g values, turn flaps up, begin flight	+	+	+	+	+	+	+
10	" " " down, " "		+	+	+	+	+	+
11	" " flaps up, end flight			+	+		+	+
12	" " " down, " "			+	+		+	+
13	g values, symmetrical, flaps up, V+H variation				+		+	+
14	" " " down, " "				+			+
15	g values, turn flaps up, V+H variation		+		+		+	+
16	" " " down, " "				+		+	+
17	g values, symmetrical, flaps up } begin flight after fuel transfer							+
18	" " " down }							+
19	g values, turn flaps up }							+
20	" " " down }							+
21	elevator, flaps up, begin flight			+	+	+	+	+
22	" " " down, " "			+	+		+	
23	" " flaps up, end flight	+	+		+		+	+
24	" " " down, " "		+		+		+	
25	rudder, , flaps up, begin flight			+	+	+	+	+
26	" " " down, " "			+	+		+	
27	" " flaps up, end flight	+	+		+		+	+
28	" " " down, " "		+		+		+	
29	aileron, , flaps up, begin flight			+	+	+	+	+
30	" " " down, " "			+	+		+	
31	" " flaps up, end flight	+	+		+		+	+
32	" " " down, " "		+		+		+	
33	speedbrake, , flaps up, begin flight			+	+	+	+	+
34	" " " down, " "			+	+		+	
35	" " flaps up, end flight	+	+		+		+	+
36	" " " down, " "		+		+		+	
37	man-flaps, , begin flight			+	+	+	+	+
38	" " end "	+	+		+		+	+
39	landing gear, , flaps down, begin flight	+	+	+	+	+	+	+
40	" " " " end "	+	+	+	+	+	+	+
41	elevator, flaps up, V+H variation							+
42	" " " down, " "							+
43	rudder, , flaps up, V+H variation							+
44	" " " down, " "							+
45	aileron, , flaps up, V+H variation							+
46	" " " down, " "							+
47	speedbrake, , flaps up, V+H variation							+
48	" " " down, " "							+
49	man-flaps, , V+H variation							+
50	taxi, before flight	+	+	+	+	+	+	+
51	" after "	+	+	+	+	+	+	+
52	take off, climb till nav. speed	+	+	+	+	+	+	+
53	approach, landing	+	+	+	+	+	+	+
54	full stop (+ drag chute)	+	+	+	+	+	+	+
55	climb (+ AB)		+					
56	descend (+ speedbrake)		+					
57	machrun, 1g, 45000 - 12000 ft		+					
58	dynamic overswing, 175 and 400 KIAS						+	
59	side slip, 175 and 400 KIAS (+ reverse)	+	+	+	+	+	+	+
60	aileron roll, 1 and 4 g	+	+	+	+	+	+	+
61	rudder roll, 1 and 4 g		+	+	+	+	+	+
62	looping, 4 g		+	+			+	
63	1 g stall, flaps up	+	+	+	+	+	+	+
64	" " down			+	+		+	+
65	inverted flight, -1 g		+		+		+	

TABLE 5

Overview of measurements in flight program (continued)

nr.	measurement \ configuration code / configuration	1	2	3	4	5	6	7
66	glide bombing, 400 and 480 KIAS, 5 and 10°				+			
67	rocketry, 400 and 480 KIAS, 20 and 30°				+			
68	dive bombing, 400 and 480 KIAS, 45°				+	+		
69	store separation					+		
70	negative g escape, 25000 ft, 420 KIAS		+	+	+		+	
71	diving spiral, 2 circles, 420 KIAS		+	+	+		+	
72	vertical reverse, ± 70 KIAS in top		+				+	
73	high speed yoyo, 10000 ft						+	
74	split S (+ AB), 7 g, 15000 ft		+	+	+		+	
75	high g barrel, 400 KIAS (+ max G)		+	+	+		+	
76	break manoeuvre (+ AB), max C_L		+	+	+		+	
77	high g barrel 250 KIAS						+	
78	low speed yoyo						+	
79	part ACT, offensive, defensive, with NF-5A		+					
80	part yet wash		+					
81	part turbulence			+		+		

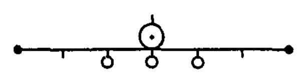
Remark: measurements 3-12 and 17-40 made at 10000 ft with 400 KIAS

TABLE 6

Example of detailed program for a flight

Flight $\boxed{5}$ in flight program

Configuration $\boxed{4}$



Stores: 3x 150 gallon tanks

exercise/ measurement	G g	V KIAS	M -	H feet	flap up ↑ down ↓	run nr.	Remarks
groundrun, empty tiptanks						2	bef. ? flight
groundrun, full tiptanks						3	before flight
cruise	0/1	400		10000	↑	4-5	
side slip + reverse	1	175		10000	↓	6	
side slip + reverse	1	400		10000	↑	7	
aileron rol'	1/4	400		10000	-	8-9	
rudder roll	1/4	400		10000	-	10-11	
inverted flight	-1	400		10000	-	12	
glide bombing	-	400		-	-	13,15	at range, 5 and 10° | from
glide bombing	-	480		-	-	14,16	at range, 5 and 10° | pitch
rocketry	-	400		-	-	17,19	at range, 20 and 30° | up
rocketry	-	480		-	-	21,20	at range, 20 and 30° | thru
dive bombing	-	400		-	-	22	at range, 45° | reco-
dive bombing	-	480		-	-	23	at range, 45° | very
cruise	0/1	400		10000	↑	?24-25	
g values, symmetrical	-	500		1000	↑	26-28	-0.5 1.0 3.0 -
g values, turn	-	500		1000	↑	29	- - 3.0 -
g values, symmetrical	-	400		1000	↑	30-32	-0.5 1.0 3.0 -
g values, turn	-	400		1000	↑	33	- - 3.0 -
g values, symmetrical	-	400		1000	↓	34-36	-0.5 1.0 3.0 -
g values, turn	-	400		1000	↓	37	- - 3.0 -
g values, symmetrical	-	500		1000	↑	38	5.0
g values, turn	-	500		1000	↑	39	5.0
groundrun, empty tiptanks						40	after flight
groundrun, full tiptanks						41	after flight

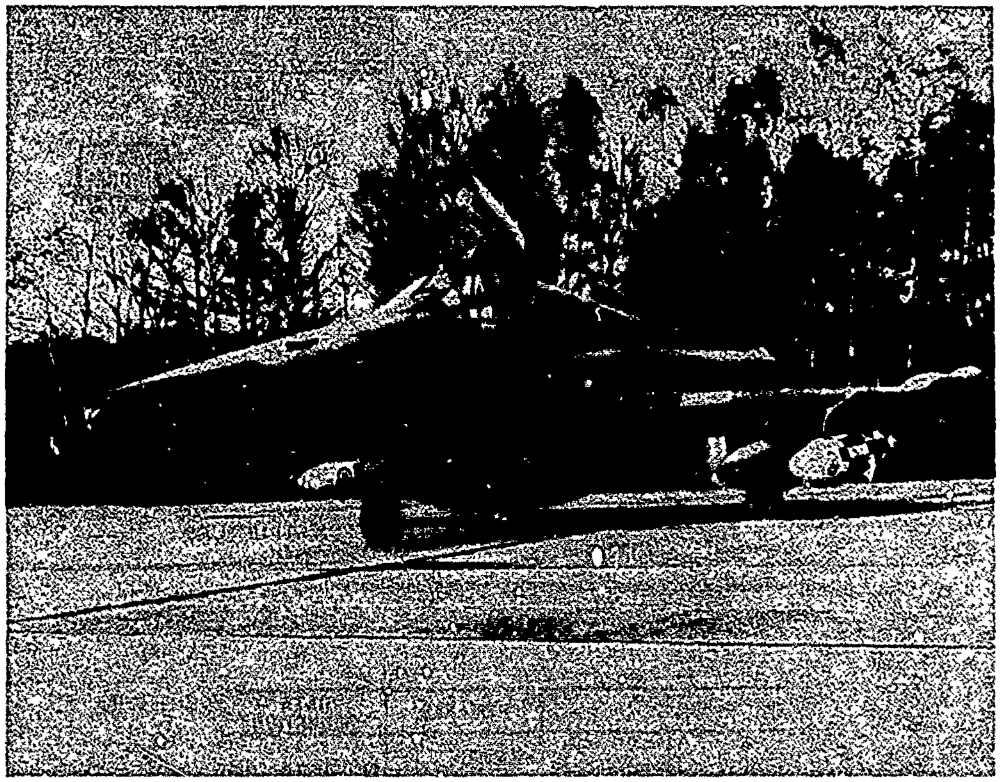


Fig. 1 NF-5A test aircraft with instrumented wing

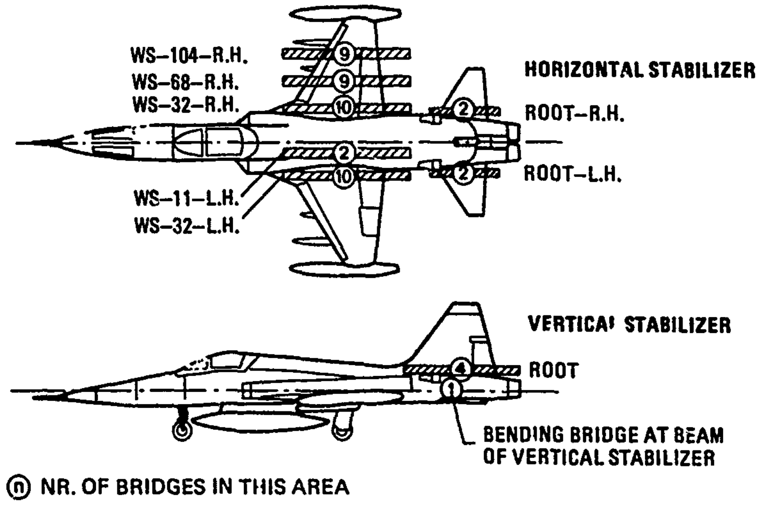


Fig. 2 Locations of straingage bridges

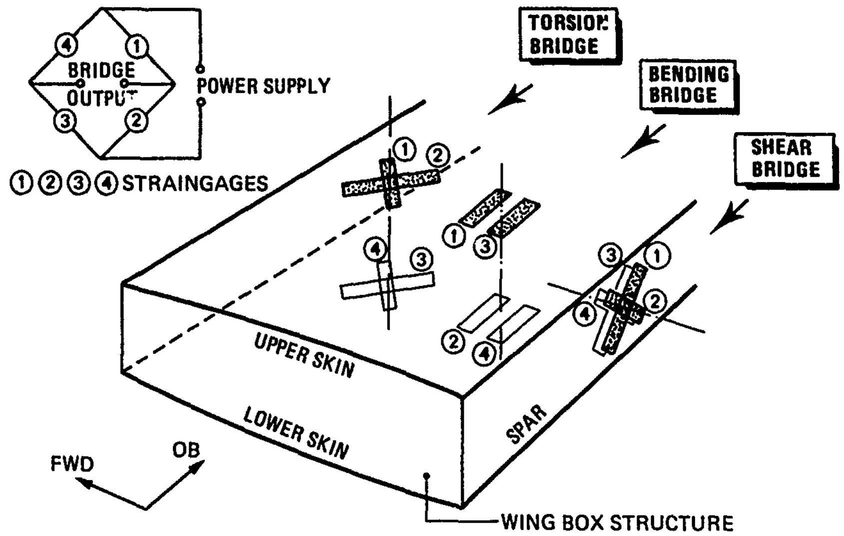


Fig. 3 Overview of straingage bridge configurations in wing

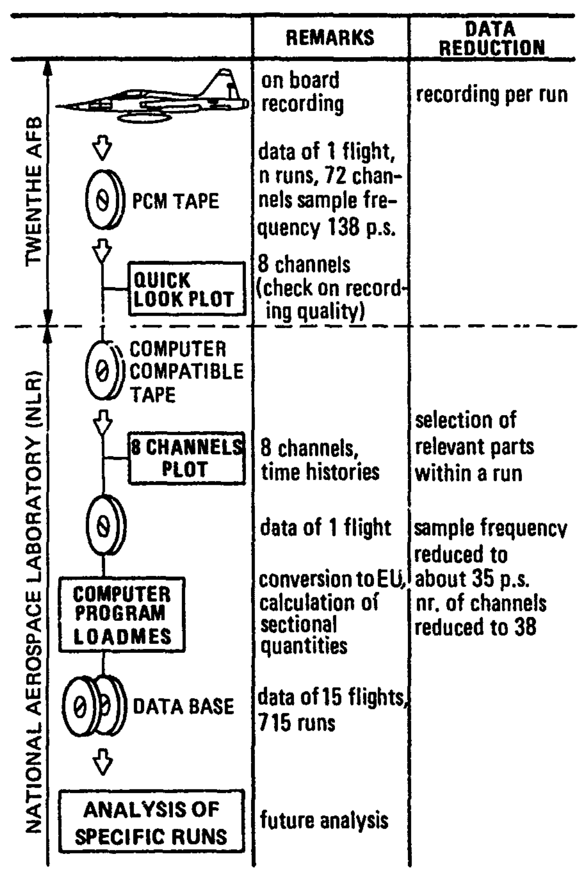


Fig. 4 Data flow

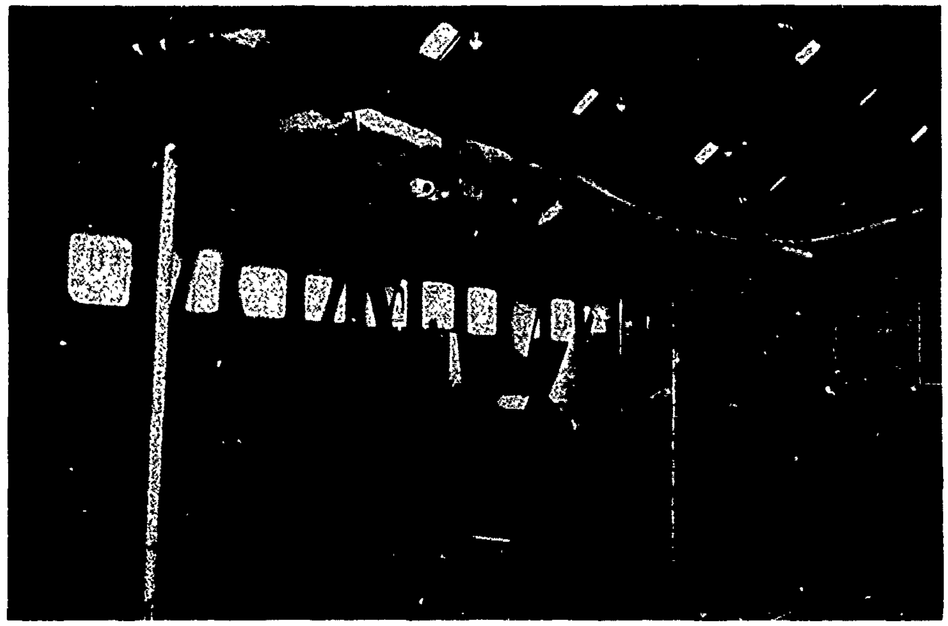


Fig. 5 Calibration set up of the wing

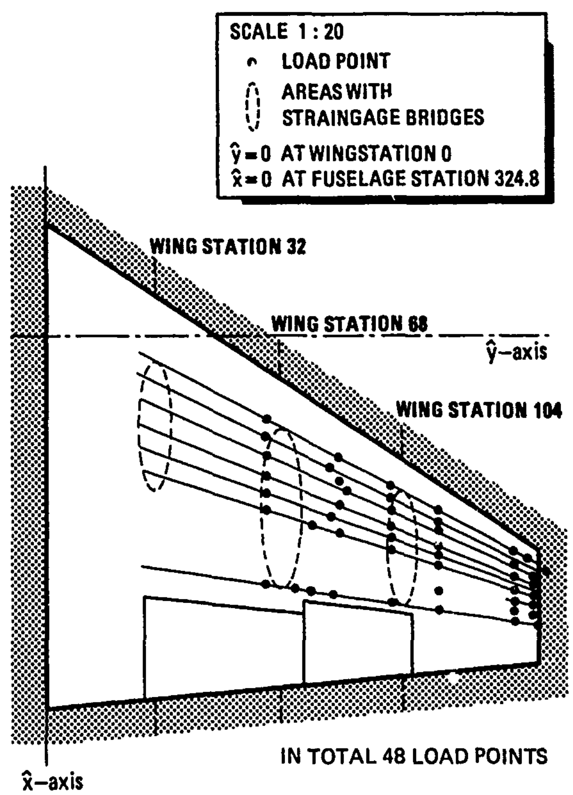


Fig. 6 Locations of the load points on the wing (also on L.H. wing)

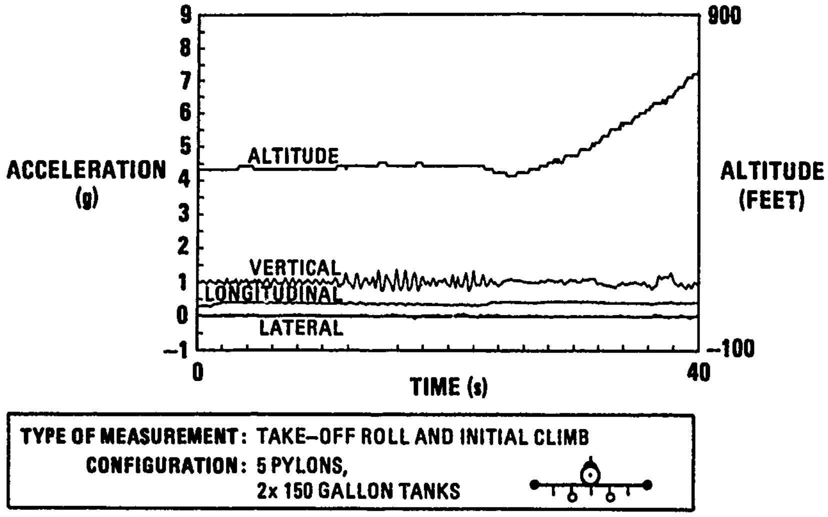


Fig. 7 Example of graphical presentation

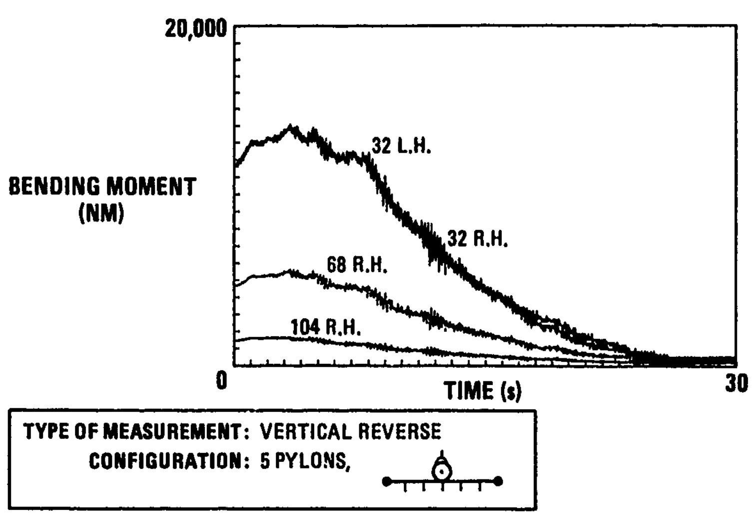


Fig. 8 Example of graphical presentation.
(Bending moment time histories at different wing stations)

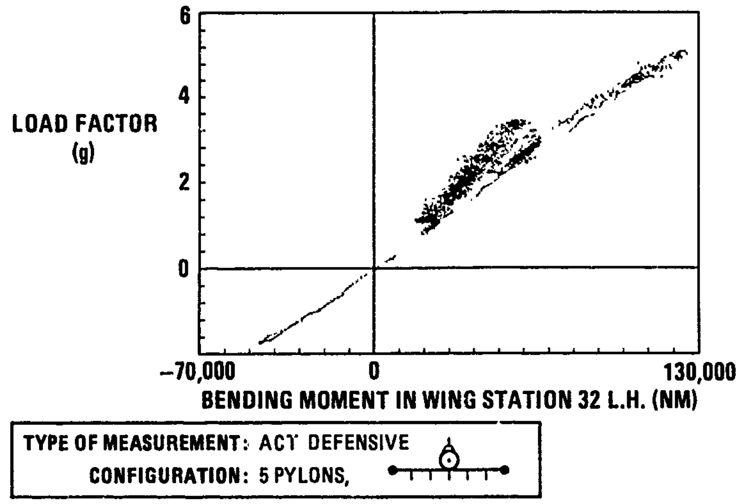


Fig. 9 Example of graphical presentation.
(Load factor versus bending moment)

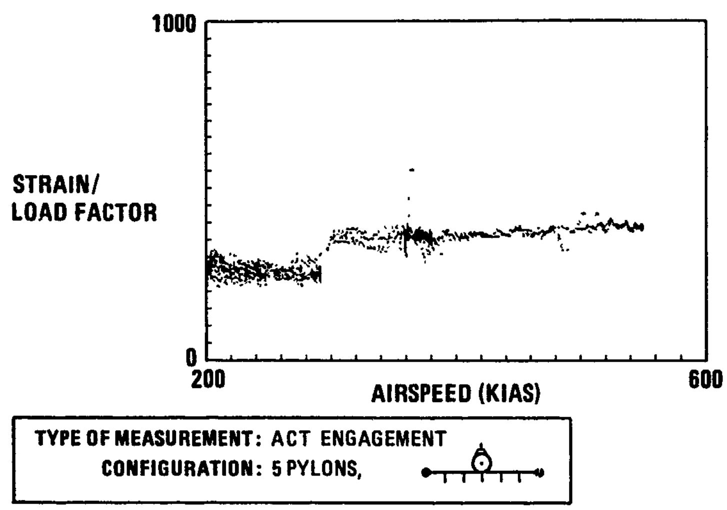


Fig. 10 Example of graphical presenta*ion.
(Strain/g versus speed)

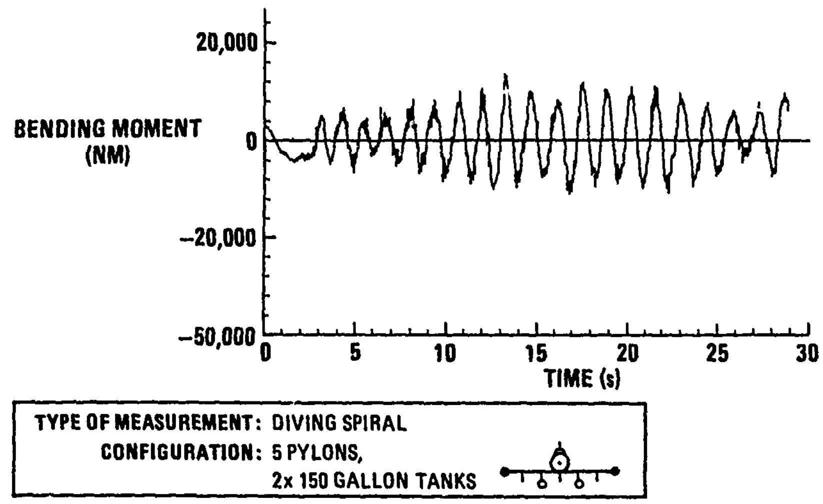


Fig. 11 Time history of fin bending moment

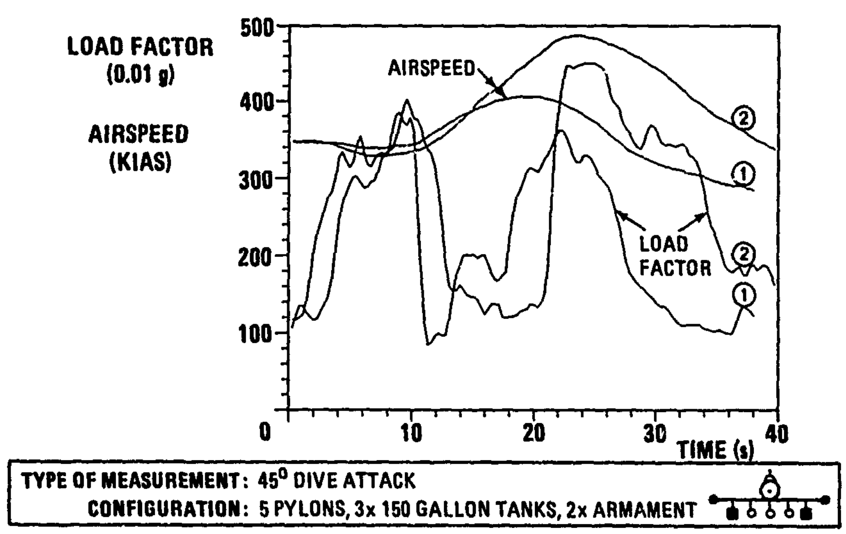


Fig. 12 Time histories for two different airspeeds

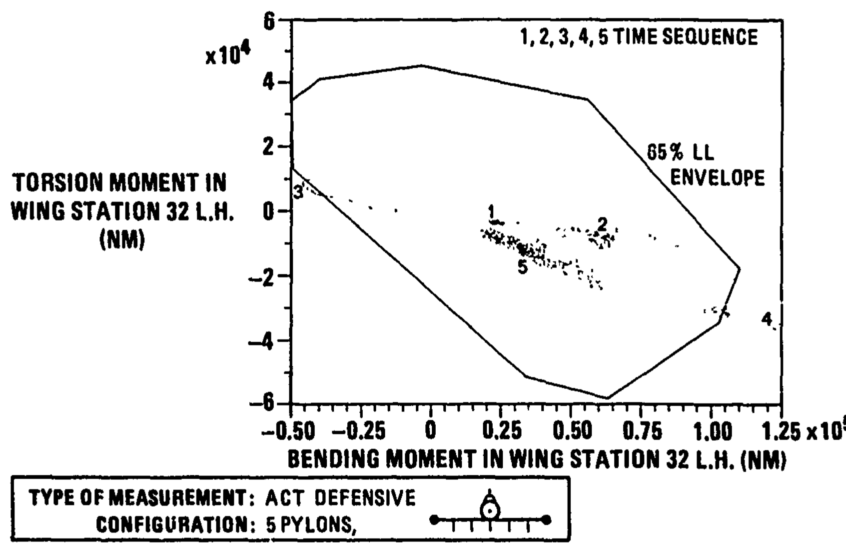


Fig. 13 Bending/torsion moment relation variation during one run

NAVY OPERATIONAL LOADS DATA SOURCES AND SYSTEMS

A. H. JOHNSON
Airframe Engineering Branch Head (Code 6042)
Naval Air Development Center
Warminster, PA 18974, USA

M. J. DUBBERLY
Structures Branch Head (Code 5302)
Naval Air Systems Command
Department of the Navy
Washington, DC 20361, USA

SUMMARY

The counting accelerometer, together with Fleet utilization data and load surveys, has provided the U.S. Navy with a means of monitoring fatigue life for most of the current Fleet aircraft. The more complex structural design features of the newer aircraft such as the F-14 and F-18 with wider fuselages, tail augmented roll, swing wings, and computerized flight control systems, present fatigue monitoring requirements that cannot be accommodated solely with a single parameter or counting accelerometer monitoring system. This paper will:

1. Provide an overview of the Navy's total Aircraft Structural Life Surveillance Program.
2. Present the current Fatigue Monitoring Program operational achievements and costs.
3. Describe the loads data acquisition systems for the F-14, F-18, A-7, and A-3 aircraft.
4. Describe the tracking programs, results to date, and discuss future expectations regarding operational features, results, and costs.

Operational loads data from the Tactical Air Combat Training System for air combat training, and the 70 mm film system for landing loads, will be described.

LIST OF SYMBOLS AND ACRONYMS

N_z - Normal acceleration load factor at aircraft center-of-gravity (CG).

G - Unit of acceleration equal to earth's gravity.

CG - Center of gravity.

G.W. - Gross weight.

FCLP - Field carrier landing practice.

TG&B - Touch, go, and bolter.

MSDRS - Maintenance Signal Data Recorder Set.

V - Airspeed.

H - Altitude.

F - Fuel State.

ASLS - Aircraft Structural Life Surveillance.

SAFE - Structural Appraisal of Fatigue Effects.

SLAP - Service Life Assessment Program.

SLEP - Service Life Extension Program.

TACTS - Tactical Aircrew Combat Training System.

EMS - Engine Monitoring System.

INTRODUCTION

This paper provides an overview of the U.S. Navy Aircraft Structural Life Surveillance Program (ASLS) and the role that operational aircraft loads data acquisition systems play in monitoring the structural integrity of the Fleet aircraft. The focus will be on systems that are designed to provide loads spectrum information and individual aircraft fatigue life tracking for the Navy's Fleet aircraft. The Fleet is currently composed of a wide range of aircraft types of varying ages and structural complexity. The loads data acquisition systems also vary in complexity from the counting accelerometers to multiparameter microprocessor controlled recorders.

The U.S. Navy's operational aircraft fleetwide structural life surveillance and management efforts are carried out by the Naval Air System Command's ASLS Program. This program is composed of three separate subprograms which are:

1. Structural Appraisal of Fatigue Effects (SAFE).

2. Service Life Assessment Program (SLAP).

3. Service Life Extension Program (SLEP).

The SAFE Program is conducted by the Naval Air Development Center. The primary purpose of the SAFE Program is to provide individual aircraft tracking to maintain current fatigue life status for all of the Navy's fixed wing operational aircraft. Efforts are currently underway to expand the SAFE Program to include helicopters. The initial efforts, addressing the SH-60 and CH-53, are expected to provide multiparameter tracking system concepts that will enable fatigue life of the airframe and dynamic system components to be tracked. The current SAFE Programs are predominantly based on counting accelerometer systems which are not bad for tracking wing structure, but they don't provide a direct measure of:

1. Effects of altitude and airspeed (especially on modern aircraft). Figure 1 illustrates the pronounced effect that airspeed and altitude have on the F-18 wing root bending moment.

2. Effects of aircraft weight on fatigue damage.

3. Effects of load sequence.

4. Effects of gusts on gust sensitive structure.

5. Fatigue damage on structure other than the wing.

These effects are currently accounted for by SLAP Program efforts such as flight and ground loads surveys conducted in the operational environment.

The SLAP Program's primary purpose is to provide an accurate picture and evaluation of what the aircraft are actually doing. The efforts consist of engineering studies, full scale fatigue tests, and flight and ground loads spectrum surveillance work needed to address current structural integrity issues which generally are to corroborate design assumptions and supplement results of the individual aircraft fatigue life tracking programs. Typical SLAP activity would be operational flight loads surveys to verify or update assumptions made to obtain aircraft fatigue life status using single parameter data to measure usage severity, followed by fatigue analysis and/or testing to evaluate the airframe in its measured environment.

The SLEP Program efforts are directed at keeping old aircraft around longer. These efforts generally consist of major structural integrity analysis, test, and modification efforts that are needed specifically to extend a known short life. Thus, SLEP assures structural integrity for older Fleet aircraft that need to be operated beyond their original design life. Navy policy normally doesn't allow any aircraft to operate beyond the fatigue life substantiated by full scale test. Since the Navy designs and tests to critical points-in-the-sky with load frequencies that are commensurate with the worst case aircraft, it is not unusual to have a reserve of structural life left after the test-substantiated flight hour limits have been reached. For example, a review of the Navy tactical aircraft has shown that the "average" Fleet aircraft, in terms of severity of usage/flight hour, will last approximately twice the test demonstrated flight hours before it reaches 100% fatigue life expended. When expressed in terms of system life cycle costs, this represents an enormous savings achieved through the individual aircraft tracking (e.g. SAFE) program. It is the result of this SAFE Program, supplemented by SLAP efforts, that provides the baseline from which needed SLEP structural modifications will be scheduled and implemented.

A profound example of application and results of all these programs is the Navy's aged A-3 aircraft which has been subjected to extensive SLEP activity to allow extension from an original design life of 3,000 hours to 18,000 flight hours. This newly substantiated 18,000 hour life, coupled with a new and improved fatigue life tracking program, is expected to allow continued operation of the A-3 into the mid-1990's.

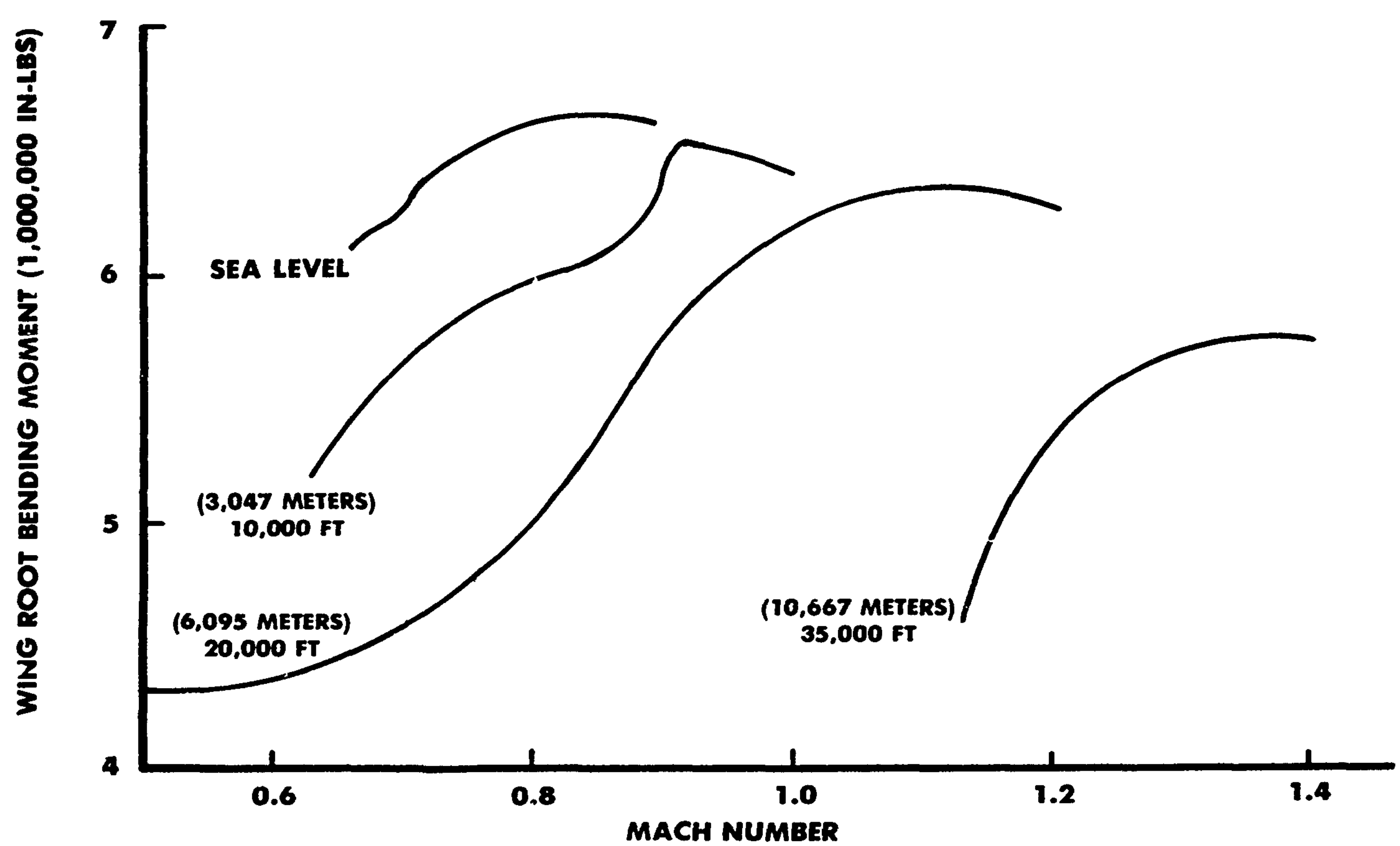


Figure 1. WING ROOT BENDING MOMENT
$N_Z = 7.5$
G.W. = 32,357 LBS

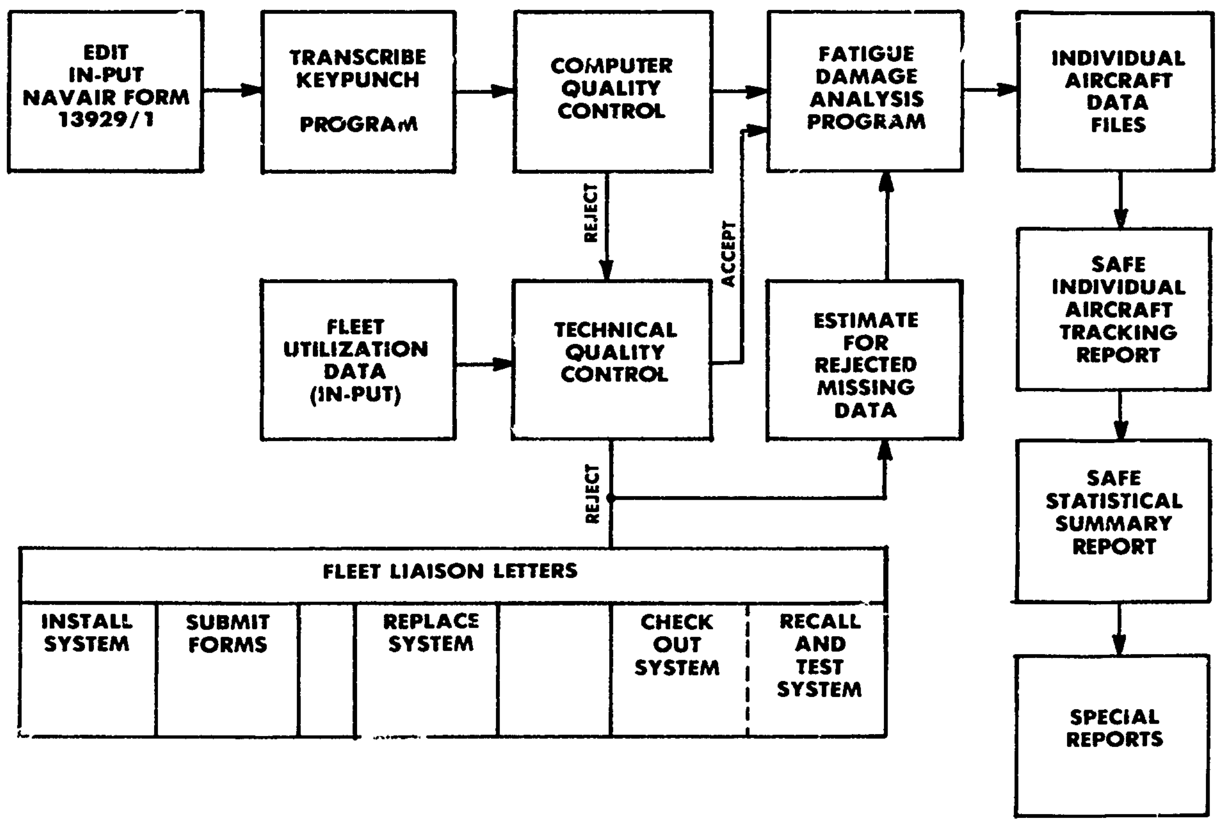


Figure 2. SAFE PROGRAM OPERATIONAL FLOW CHART

DISCUSSION OF THE CURRENT SAFE PROGRAM

The current fleetwide fatigue life tracking program (e.g. SAFE) will be described in more detail to provide a base line for assessing the new systems and also to illustrate the operational features that will carry over to these new systems. The SAFE Program provides individual fatigue life expended values and rates for approximately 3,500 of the Navy's operation l fixed wing aircraft consisting of about 55 different model configurations with differing fatigue requirements and concerns. These differences are generally accommodated by the fatigue damage methodology and routines; however, some models do require unique variations in input, output, and reporting requirements. The program results consist of fatigue life expended rates, status, and other life limiting data for all aircraft which are compiled and published quarterly in a single report. A second report providing statistical evaluation results of measured usage severity is published twice yearly.

The SAFE Program typically relies on counting accelerometer data to assess flight maneuver damage which is combined with other Fleet utilization data to establish ground-air-ground, catapult, arrestment, and other life limiting factors needed to determine fatigue life status. Figure 2 provides a flow chart that depicts the basic operations involved in running the SAFE Program. The usage severity input data is received on NAVAIR Form 13920/1, which is depicted in Figure 3. This is a postcard-size form that is used to manually record and transmit individual aircraft counting accelerometer readings to the Naval Air Development Center for processing. At 100% efficiency, this would amount to one card per month for each Fleet aircraft. This data is manually edited and transcribed via a keypunch process to an automated quality control program. The quality control program evaluates the data for reasonableness and flags any data that varies from expected limits. The flagged data is manually accepted, or corrected and accepted, or totally rejected. The rejected and missing data is replaced with estimated data. The counting accelerometer data is merged with the Fleet utilization data for individual aircraft fatigue life expended and status determination. This computerized process provides the tables needed for the SAFE quarterly report. The content and format for a typical SAFE report result is depicted in Figure 4.

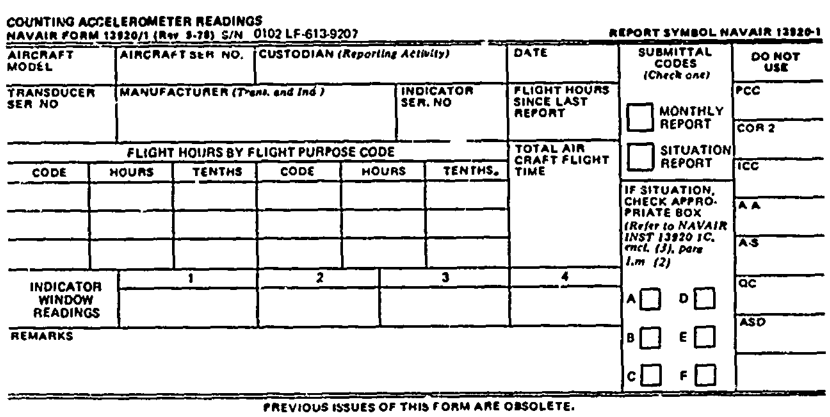

COUNTING ACCELEROMETER READINGS
NAVAIR FORM 13920/1 (Rev 9-78) S/N 0102 LF-613-9207
REPORT SYMBOL NAVAIR 13920-1
AIRCRAFT MODEL
AIRCRAFT SER NO.
CUSTODIAN (Reporting Activity)
DATE
SUBMITTAL CODES (Check one)
DO NOT USE
TRANSDUCER SER NO
MANUFACTURER (Trans. and Ind.)
INDICATOR SER. NO
FLIGHT HOURS SINCE LAST REPORT
MONTHLY REPORT
PCC
COR 2
FLIGHT HOURS BY FLIGHT PURPOSE CODE
CODE | HOURS | TENTHS | CODE | HOURS | TENTHS.
TOTAL AIR CRAFT FLIGHT TIME
SITUATION REPORT
ICC
IF SITUATION, CHECK APPROPRIATE BOX (Refer to NAVAIR INST 13920 1C, encl. (3), para l.m (2)
A A
A S
INDICATOR WINDOW READINGS
1 | 2 | 3 | 4
QC
A D
ASD
REMARKS
B E
C F
PREVIOUS ISSUES OF THIS FORM ARE OBSOLETE.

FIGURE 3. Counting Accelerometer Data Input Form

The good data capture rate for this program has stabilized at about 75%. This relatively high capture rate can be attributed to the Fleet liaison letter feature which automatically identifies and prepares a letter request for needed corrective actions. Implementation of this feature brought the good data capture rate up from below 50% to the current 75%.

The cost of maintaining and operating the SAFE Program currently runs about \$1,000,000/year. This breaks down to about \$850,000 for labor, with the remaining \$150,000 required for materials and computer costs. The SAFE Program currently tracks about 3,500 aircraft, so this cost breaks down to \$285/aircraft/year. There are some additional costs associated with this program that must be considered if one is to use it as a basis for evaluating the new multiparameter tracking programs. These costs are for the SLAP functions such as flight and ground loads surveys needed to establish or verify the effect of assumptions that are made for the original design and when single parameter data is used for fatigue life tracking. The current expenditures for this work average about \$3,000,000/year. The total will then establish a \$4,000,000/year average cost for SAFE and SLAP work that applies to approximately 5,000 aircraft or \$800/aircraft/year. These figures will be used later to define the expected cost impact of the new more complex fatigue life tracking systems.

INDIVIDUAL AIRPLANE DATA															
							LANDING DATA							LIFE EXPENDED	
AIRCRAFT SERIAL NUMBER	SERVICE ACCEPTANCE DATE	AIRCRAFT CUSTODIAN	MONTHS IN TOUR	TOTAL FLIGHT HOURS	PERCENT TIME IN COMBAT	PERCENT ACCEPTABLE CA DATA	ARRESTMENTS	FCLP TG&B	FIELD	TOTAL	PERCENT ACCEPTABLE LDG DATA	ESTIMATED CATAPULTS	PROGRAM CODE	PERCENT	RATE IN % PER 1000 HOURS

Figure 4. STRUCTURAL FATIGUE LIFE PROGRAM REPORT

NEW STRUCTURAL APPRAISAL OF FATIGUE EFFECTS (SAFE) TRACKING PROGRAMS

Multiparameter loads data acquisition systems to provide for loads spectrum and fatigue life monitoring will be included as a requirement for all new Navy aircraft procurement and major modification programs. They will also be included as part of some life extension programs, particularly when adequacy of the existing fatigue life tracking program is in question. The new fatigue life tracking programs covered in this paper are for the F-18, F-14, A-7, and A-3 aircraft. The F-18 loads data acquisition system was included in new aircraft procurement. The F-14 system is designed for retrofit, but will also be included in the extensively modified F-14D new aircraft procurement. The A-7 program took advantage of a major Engine Monitoring System (EMS) update to piggyback loads data acquisition capability, and the A-3 loads data acquisition system is part of the Service Life Extension Program (SLEP). These aircraft, and the new fatigue life tracking programs, provide a wide range of requirements and complexity that will be illustrative of the kinds of options that are currently available. These programs when implemented will be tailored to operationally fit into the framework of the SAFE Program.

THE F-18 FATIGUE LIFE TRACKING PROGRAM

The F-18 operational loads data is obtained by the Maintenance Signal Data Recorder Set (MSDRS) installed in each aircraft during manufacture. The MSDRS is a cartridge type wire recorder that is controlled by and receives signals from the mission computer. The loads data signals consist of readings from seven strain sensors, four flight parameters, fuel state, and stores configuration. The operational loads data is only part of the information that the MSDRS records so at this point the loads data is intermingled with other data. The structural loads data is stripped from this cartridge and merged with other flight usage data by a ground station computer to produce an 8-track tape containing complete loads data. This tape is mailed to the Naval Air Development Center for loads spectrum and individual aircraft fatigue life monitoring.

The F-18 loads data includes strains measured at seven locations chosen to provide a measure of the respective components primary load. These strain gage locations are illustrated in Figure 5. The measured strains and parameters are:

- Wing Root Strain
- Wing Fold Strain
- Left Horizontal Stabilizer Strain
- Right Horizontal Stabilizer Strain
- Left Vertical Stabilizer Strain
- Right Vertical Stabilizer Strain
- Forward Fuselage Strain
- Altitude
- Airspeed (True)
- Normal Load Factor
- Roll Rate
- Stores Configuration
- Fuel Remaining

The data is compacted on the MSDRS cartridge by taking time slices at significant peaks and valleys of the strains and N_z. Significant peaks and valleys are identified by the mission computer. To qualify as significant, each event must pass prescribed threshold (level) and rise-fall (range) criteria. This results in only data that is needed to determine structural component fatigue damage being recorded. Judicious selection of the data compaction criteria is important for the production units, as low thresholds reduce efficiency, and high thresholds reduce accuracy. In spite of this, extreme caution must be used not to start out with thresholds that are too high as this will result in losing the data that is needed to establish the correct thresholds. In other words, you can't evaluate the effect of the load cycles that are being eliminated unless you know what they are. An example of the recorded loads data is shown in Figure 6.

For fatigue life tracking, this data will go through the same kind of central processing as the counting accelerometer data for the current SAFE Program. The postcard editing, keypunching, and quality controlling process used for the SAFE Program will be replaced with tape drives and computer software that will sort the incoming data to load individual aircraft files and perform quality checks on the data. Questionable or missing data will be identified for evaluation by the technicians or structural integrity engineers. The missing and bad data will be statistically replaced with representative data.

The analysis program will have the ability to determine fatigue life (crack initiation) expended and crack growth for designated points on the major structural components. The fatigue life tracking program will be capable of maintaining sequential loads data history files and structural component fatigue life expended history files. These files will be used as a basis for structural integrity monitoring and the reliability centered maintenance Age Exploration Program.

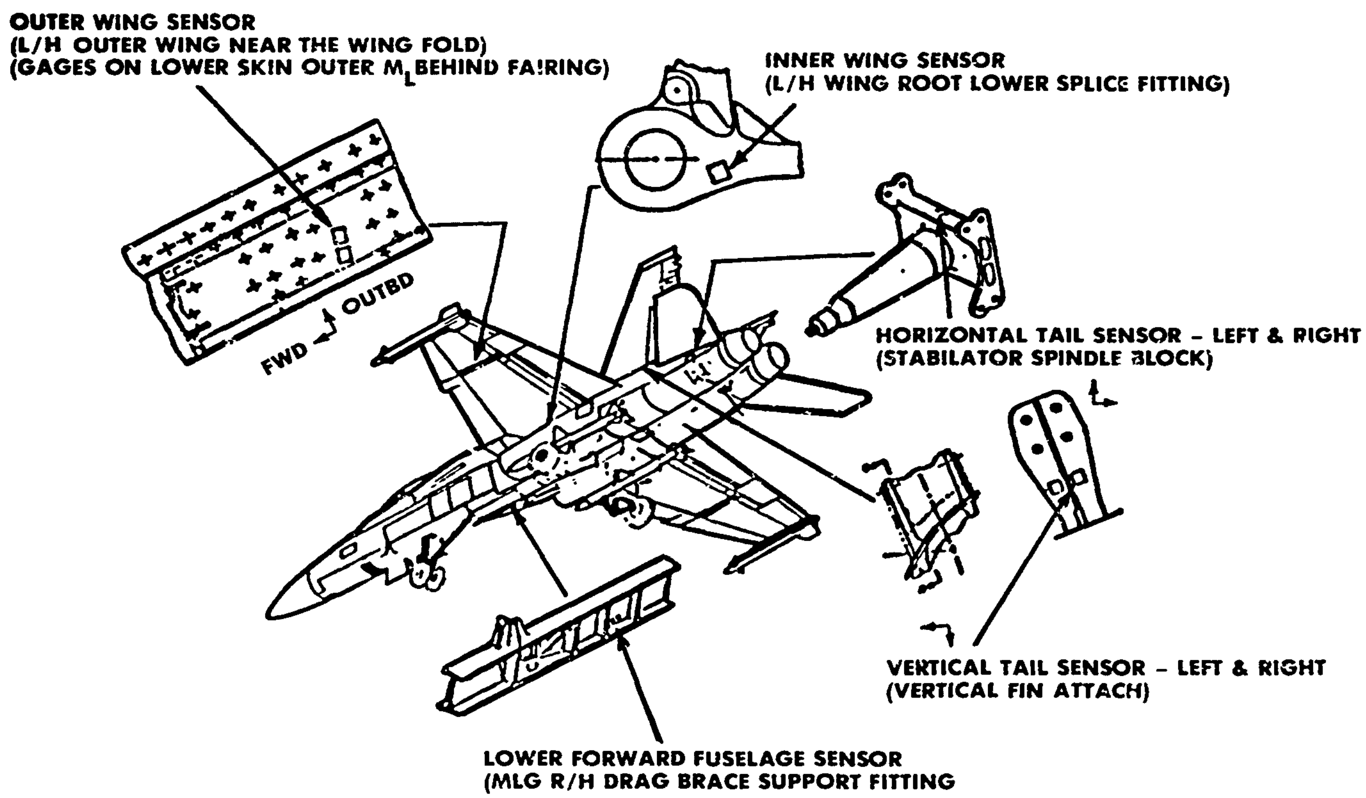


Figure 5. F/A-18 STRAIN SENSOR LOCATIONS

F A T I G U E   S E N S O R S   P E A K S   A N D   V A L L E Y S

PDTIME (SEC)	WING ROOT STRAIN (USTRN) LEFT	WING FOLD STRAIN (USTRN) LEFT	HOR TAIL STRAIN (USTRN) LT	HOR TAIL STRAIN (USTRN) RT	VERT TAIL STRAIN (USTRN) LT	VERT TAIL STRAIN (USTRN) RT	FWD FUS STRAIN (USTRN)	ELAPSED TIME (SECONDS)	FUEL QUANTITY (LBS)	A/C TAS (KNOTS)	A/C ALT (FT)	A/C ROLL (DEG/S)	A/C ACC (G)	FTIME (SEC)	CODE ID	SN
INIT VALUE	200	72	488	-1384	248	744	-640									
1198.75	32	328	360	-304	0	-96	-152	1193.75	8960	200	66	-.39	1.22	3547	7	10
1220.45	32	280	392	-328	8	-96	-144	1193.05	8992	193	58	.38	1.09	3569	11	10
1223.15	32	296	352	-344	0	-104	-136	1193.65	8992	204	64	-.52	1.20	3572	14	10
1348.85	64	40	-80	88	8	-8	-32	1264.55	8800	367	4776	.38	.66	97	8	10
1367.25	72	256	152	-160	0	16	-224	1357.65	8704	306	5029	.80	2.15	116	7	10
1430.75	80	48	-24	56	8	8	-112	1403.35	8448	390	9893	.45	.88	179	8	10
1436.15	96	224	168	-104	-16	8	-296	1432.85	8384	447	12952	.22	2.45	185	7	10
1586.95	88	16	-32	86	8	32	-112	1572.25	8288	474	20106	-7.33	.89	335	8	10
1588.65	88	96	-136	-24	24	16	-144	1537.45	8288	494	19956	22.50	1.03	337	12	10
1589.65	88	48	208	168	-8	8	-136	1440.85	8320	404	15744	-45.09	1.11	338	13	10
1595.35	112	328	224	-184	-24	24	-408	1591.15	8256	458	19931	.16	3.25	344	7	10
1596.85	112	296	280	-168	-16	24	-384	1591.35	8256	460	19941	-.70	3.07	345	11	11
1653.05	72	16	-16	32	40	32	-112	1600.75	8288	436	20407	-1.42	.73	401	8	11
1653.75	80	80	-72	-96	48	16	-152	1597.45	8160	439	19998	17.66	1.17	402	12	11
1663.35	120	400	360	-224	-16	32	-472	1655.15	8064	451	19815	-4.59	3.75	412	11	11
1665.25	96	464	-48	-104	-32	24	-384	1664.05	8000	402	19776	46.83	3.70	414	12	11
1665.35	96	496	56	-248	-32	88	-416	1662.85	8000	413	19682	49.92	3.79	414	7	11
1679.15	96	288	400	-136	32	16	-344	1665.75	7968	387	19974	3.27	2.74	428	11	11
1692.35	72	24	136	8	32	24	-136	1686.45	8000	370	19662	-17.61	.94	441	8	11
1702.25	64	88	0	-144	56	16	-176	1690.15	7936	385	19631	22.52	1.20	451	12	11
1702.85	112	400	360	-288	-16	32	-472	1656.65	8032	446	19824	-2.33	3.72	451	14	11
1704.75	104	208	552	232	56	-56	-344	1703.35	7776	401	19402	-92.69	2.58	453	13	11
1705.25	104	192	584	216	64	-72	-352	1703.25	7808	400	19408	-85.09	2.60	454	11	11
1707.35	88	352	176	-120	0	24	-392	1705.55	7808	399	19353	-25.41	3.23	456	7	11
1711.15	96	288	56	-376	8	64	-336	1706.45	7808	398	19328	20.44	2.74	460	14	11
1741.55	104	456	320	-224	-8	32	-504	1657.65	8000	441	19817	-.63	3.99	490	10	11
1887.95	88	-24	8	104	32	24	-96	1879.25	7648	491	19052	-12.30	1.00	636	8	11
1888.75	88	8	-104	16	40	16	-96	1795.95	7840	494	19268	9.92	1.02	637	12	11
1893.75	96	-8	64	152	24	24	-120	1805.45	7808	493	19261	-25.00	1.08	642	13	11
1893.75	128	280	336	-104	16	32	-416	1891.85	7616	470	19303	-11.81	3.47	642	7	11
1894.05	128	272	408	-120	8	32	-424	1891.95	7584	469	19316	-12.98	3.50	642	11	11
1917.55	[illegible]	48	-120	-72	64	8	-120	1914.15	7616	453	19443	31.53	1.09	666	12	11
1917.65	80	8	0	16	48	16	-120	1911.55	7552	449	19533	3.75	1.08	666	8	11
1919.75	120	336	360	-232	16	24	-432	1919.35	7552	455	19100	-.08	3.65	668	14	11
1919.05	120	368	268	-184	16	24	-432	1918.75	7552	457	19161	1.55	3.83	668	7	11
1921.[illegible]5	128	232	704	248	112	-112	-432	1919.85	7552	453	19045	-47.33	3.50	670	11	11
1921.95	136	232	696	256	112	-112	-432	1919.95	7552	452	19035	-59.19	3.46	670	13	11
1922.95	136	184	600	200	40	0	-392	1920.65	7648	450	18977	-84.45	3.08	671	8	11
1924.45	88	8	-64	72	12	24	-72	1780.05	7872	494	19192	-.09	.93	673	9	11
1927.65	96	464	80	-256	0	80	-440	1925.35	7520	422	18740	41.06	4.12	676	7	11
1928.35	96	408	32	-200	32	48	-392	1926.95	7520	405	18800	50.88	3.67	677	12	11
1928.[illegible]5	104	344	144	-424	8	64	-432	1923.15	7584	440	18879	9.58	3.70	677	14	11
1930.35	80	216	400	-88	64	8	-304	1920.65	7520	389	18951	.56	2.65	679	8	11
1935.15	88	216	440	-96	64	8	-328	1920.95	7552	388	18968	-17.33	2.72	682	11	11

FIGURE 6. F-18 Fatigue Life Tracking Program Loads Data

To this point, we have discussed the capabilities of the F-18 fatigue life analysis and tracking programs. Now we will delve into some of the economic issues. Projecting computer and manpower costs for operating this system for a fleet of F-18 aircraft is still somewhat tentative, but some data to base projections on is now available.

The most recent projected operating costs/flight hour for the F-18 Fatigue Life Tracking Program are .04 man-hours plus $2.00 for computer processing. Based on 300 flight hours/year/aircraft, this would require 12 man-hours plus $600/aircraft/year or about $1,080/aircraft/year. This is in excess of the $800/aircraft that is required to conduct both of the current Structural Appraisal of Fatigue Effects and Service Life Assessment Program activities. At first glance this might be discouraging, so we need to look further into the prospects for reducing these projected costs. First, the cost projections are based on current Interim Fatigue Life Tracking Program experience, adjusted to account for program improvements and operational procedures that would be employed directly on the current Naval Air Development Center's CDC 6600/Cyber 75 computer system. Preliminary investigations indicate that the computer costs could be significantly reduced by employing a dedicated minicomputer to accomplish most of the Fatigue Life Tracking Program functions. It is also expected that new inovations in computer equipment and procedures will reduce the projected man-hour costs. Since these programs are inherently phased into operation by the production schedule, there will be opportunities to evaluate cost reduction features using real data before the aircraft numbers result in overwhelming costs. In this regard, the F-18 Fatigue Life Tracking Program will lead the way for other U.S. Navy aircraft multiparameter fatigue life tracking programs.

Another characteristic of cost projections for these new systems is a near term bulge in effort during the transition period to new or modified aircraft. The bulge is due to the fact that need for SLAP activity will not subside for sometime because it focuses on older Fleet aircraft. It will take several years before the results from these new more complex tracking programs begin to change the SLAP activity. Additionally, one cannot expect that future SLAP activity will be significantly reduced by these new programs. More likely, there will be an increase in analytical requirements to improve Fleet aircraft structural integrity at a modest increase in cost. These cost increases will be more than offset by the tremendous savings that can be achieved through: improved design criteria, early recognition and resolution impending structural problems, and the full utilization of aircraft structural life; all of which will result from information provided by the new tracking programs. The beneficial effect on flight safety should also be considered. To put this in context, just consider that the cost associated with procurement of one aircraft is more than the total development and projected operating costs for 20 years of fatigue life tracking for 1,500 F-18 aircraft.

In order to provide the capability to reconstruct the load history for evaluation of unexpected failures, the individual aircraft load history data files must be maintained indefinitely. These files plus additional files needed to fill in for bad or missing data and other related operations combine to form substantial data storage requirements. Let's just look in detail at the loads data history files for now. If each triggering parameter (seven altogether) results in an average of 50 time slices/flight hour and each time slice consists of two records (peak and valley) of 16 words, you will get 2 x 7 x 50 x 16 words per flight hour. This results in 11,200 words per hour for a 6,000 hour life or 67,200,000 words. This would indicate a need for about 10 to 12 nine-track computer tapes for each airplane to store its lifetime of loads data in sequential form. Projecting out-year data storage requirements of this magnitude for a large number of Fleet aircraft could result in up to 3,000 aircraft in various stages of their lives with an average of five tapes each, or a total of 15,000 tapes. This could probably be managed if it was essential to do it this way. The current Navy plans are to start collecting and saving the flight loads data in this manner until enough real data for each sensor is available to evaluate data compression and other storage concepts. The remaining data files required to provide for loads spectrum evaluations, structural component fatigue life expenditure history, and bad or missing data gap filling should fit on a single disc pack.

F-18 INTERIM FATIGUE LIFE TRACKING PROGRAM

The purpose of the F-18 Interim Fatigue Life Tracking Program is to provide Fleet aircraft fatigue monitoring until the production tracking program becomes operational. As of January 1984, 12,800 hours of data had been processed covering 48 TF- and F/A-18 aircraft. A reasonable data capture rate of 53% was obtained during this initial operating period.

The Interim Fatigue Life Tracking Program reports provide the fatigue index at the most critical location for 90% of the aircraft and fatigue indexes at all seven locations for 10% of the aircraft. Figure 7 provides a typical result from the January 1984 report. Although the small amount of data available for each aircraft could be misleading, this figure indicates that the same location does not yield the highest fatigue index for all the aircraft.

BUNO	CUM A/C	ACTIVITY	FLIGHT HOURS	PROCESSED DATA HRS.	% PROCESSED STRAIN DATA	FATIGUE INDEX	INDEX RATE/1000 HR	SENSOR CODE	AS OF DATE
161359	F19	VFA-125	352	198	56	.010	.028	W Root	01/11/83
						.006	.017	W Fold	
						0	0	Fwd Fuse	
						.006	.017	L/H Tail	
						.005	.014	R/H Tail	
						.003	.009	L/V Tail	
						.011	.031	R/V Tail	
161360	TF9	VFA-125	347	209	60	.009	.026	W Root	02/22/83
161361	F20	VFA-125	389	131	34	.010	.026	W Root	02/11/83
161362	F21	VFA-125	310	188	61	.003	.010	W Root	12/03/82
161363	F22	VFA-125	231	157	68	.005	.022	W Root	02/16/83
161364	F23	VFA-125	213	131	62	.011	.052	W Root	02/23/83
161365	F24	VFA-125	211	130	62	.011	.052	W Root	01/28/83
161366	F25	VX-5	211	172	82	.007	.033	W Fold	12/13/82
161367	F26	VFA-125	305	153	50	.002	.007	W Root	02/18/83
161519	F27	VFA-125	201	102	51	.006	.030	W Root	02/24/83
161520	F28	VX-4	278	124	45	.004	.014	W Root	02/07/83
161521	F29	VX-4	284	162	57	.007	.025	W Root	02/16/83
						.010	.035	W Fold	
						.006	.021	Fwd Fuse	
						.005	.018	L/H Tail	
						.008	.028	R/H Tail	
						.001	.004	L/V Tail	
						.004	.014	R/V Tail	

FIGURE 7. F-18 Interim Fatigue Life Tracking Program Result

F-14 FATIGUE MONITORING SYSTEM

The F-14 Fatigue Monitoring System will utilize an airborne data acquisition system that monitors and records engine and kinematic flight parameters, aircraft configuration discretes, and fuel state. This parametric data will be recorded in time slice compacted form in much the same manner as the F-18 data. The parametric signals are obtained from existing sources except for normal acceleration (N_Z) which will be obtained from an independent transducer. Since this system will be retrofit to the existing F-14 aircraft, strain sensors were not used. The recorder is a microprocessor controlled solid state memory unit with 128,000 bytes of storage capacity. The time slice recording is triggered by key parameters that experience peaks or valleys that exceed prescribed threshold limits. The solid state memory is sufficient to retain data from about one month of active flying. The recorded data is down-loaded using an electronic interrogator that also conducts health checks on the airborne recorder. The ground station concepts for this system have not been completely definitized yet, but will probably consist of feeding the interrogated loads data into a micro or minicomputer for processing and transfer to a disc or tape for mailing to the Naval Air Development Center. Table 1 provides a list of the parameters that will be monitored and recorded by this system.

The fatigue life tracking analysis software will convert the peak/valley parameter data to component loads and stresses using algorithms that will be developed by employing regression analysis to tune classical loads relationships to instrumented flight test data.

The operational characteristics of this fatigue life tracking program are expected to be very similar to those discussed for the F-18 program.

TABLE 1. F-14 FATIGUE MONITORING SYSTEM RECORDED PARAMETERS

Loads Parameters

1. CG Normal Acceleration
2. CG Lateral Acceleration
3. Roll Rate
4. Mach Number
5. Pressure Altitude
6. True Angle of Attack
7. Rudder Position
8. Left Stabilizer Position
9. Right Stabilizer Position
10. Pitch Rate
11. Yaw Rate
12. Wing Sweep (Back-Up)
13. Fuel Remaining
14. Stores

Engine Monitoring Parameters

1.	Engine No. 1 Turbine Speed	N1
2.	Engine No. 1 Turbine Speed	N2
3.	Engine No. 1 Temperature	T5
4.	Engine No. 2 Turbine Speed	N1
5.	Engine No. 2 Turbine Speed	N2
6.	Engine No. 2 Temperature	T5

Discrete Configuration Parameter List

1. Wing Sweep Manual Aft Select
2. Wing Sweep Auto Mode Select
3. Maneuver Flap Extended
4. Nose Landing Gear Launch Bar Extended
5. Nose Landing Gear Kneel Solenoid Engaged
6. Main Landing Gear Down and Locked
7. Arresting Hook Down

A-7E FATIGUE LIFE TRACKING PROGRAM

The new A-7E Fatigue Life Tracking Program will utilize the TF-41 Engine Monitoring System (EMS) for data acquisition. This program has developed because of an opportunity to economically obtain an improved fatigue life tracking system by taking advantage of an already scheduled update of the EMS. The loads data acquisition required little more than software changes as all of the needed hardware already existed. Algorithms had to be programmed into the onboard computer to again recognize and record the loads data parameters at peaks and valleys of normal acceleration that exceed the prescribed threshold limits. The loads parameters recorded are:

N_Z - Normal acceleration at aircraft CG

V - Airspeed

H - Altitude

F - Fuel State

Wing stresses will be calculated at two critical locations on the lower skin using regression relationships. These peak and valley stresses will then be used to determine fatigue life expended values. Although relatively simple, this program will remove the conservative point-in-the-sky and weight assumptions inherent in the current fatigue life tracking program. It also provides for sequence accountable crack initiation or crack growth analysis methodologies.

A significant aspect of this approach is the establishment of an improved fatigue life tracking program with very low development and implementation costs. This program has moved from the proposal stage through a demonstration program in just over a year and is expected to be in a phased implementation process by mid-1985. Figure 8 is a sample of the data from the demonstration program that illustrates the kind of information that will be provided for individual aircraft loads spectrum and fatigue life monitoring.

AIRFRAME CYCLES DATA

A/C BUNO 156851
ENG SN 141620
FLT NO.102
DATE 3306
FLT TIME 1J1

"G"	MACH NUMBER	ALTITUDE	FUEL USED	AIRCRAFT WEIGHT
0.000	0.00.	-1000.	0.	38000.
2.354	0.71	19661.	432.	37568.
0.693	0.82	19929.	432.	37568.
2.749	0.78	18186.	560.	37440.
0.303	0.69	20659.	564.	37436.
2.363	0.72	19661.	620.	37380.
0.762	0.57	19720.	632.	37368.
3.315	0.55	13777.	676.	37324.
0.80[illegible]	0.44	11230.	692.	37308.
3.862	0.53	8608.	712.	37288.
0.757	0.69	4824.	772.	37228.
1.885	0.50	4660.	876.	37124.
0.854	0.47	6388.	876.	37124.
2.485	0.49	9889.	880.	37120.
0.552	0.64	6284.	884.	37116.
4.976	0.66	3513.	888.	37112.
0.381	0.37	10098.	888.	37112.
3.369	0.48	8459.	888.	37112.
0.913	0.68	4422.	892.	37108.
5.459	0.70	2307.	896.	37104.
0.562	0.43	7684.	896.	37104.

FIGURE 8. A-7E Fatigue Life Tracking Program - Data Sample

A-3 FATIGUE LIFE TRACKING PROGRAM

The A-3 aircraft have recently completed a full scale airframe fatigue test to identify airframe changes needed to extend the service life into the 1990's. In order to maximize fatigue life utilization from individual aircraft and to schedule the downstream modifications, a new individual aircraft Fatigue Life Tracking Program was required.

The mission for the A-3 aircraft requires little high G maneuvering. The majority of the fatigue damage results from ground-air-ground cycles and gusts. The situation is also one that requires a kit type retrofit of the Fleet aircraft. In order to accomplish this at a reasonable cost, the airborne data acquisition system had to be relatively simple. The concept chosen was one that would use a 2-parameter system using a wing strain and load factor (N_Z). Examination of the aircraft revealed an accessible location inside the center wing section that was suitable for installation of a microprocessor type recording unit. This location was near the center of gravity and also the fatigue critical points on the wing. Since the recorder and sensors are in close proximity, the retrofit should be relatively easy. The data is recorded separately for flight and ground conditions in peak and range type exceedance matrices with three divisions per G resolution. This low cost concept will also be considered for improved fatigue life tracking of other low G gust sensitive Fleet aircraft where both the need exists and retrofit is required.

The down-loading will be accomplished using an electronic interrogator that will copy the flight data into a solid state memory device for mailing to the Naval Air Development Center.

The Fatigue Life Tracking Program will be capable of calculating fatigue (crack initiation) indexes and crack growth at multiple locations.

TACTICAL AIRCREW COMBAT TRAINING SYSTEM

The Tactical Aircrew Combat Training System (TACTS) was developed to improve fighter pilot performance. The system provides an instrumented airspace approximately 60-80 km in diameter by 15-20 km in altitude. The instrumentation consists of a pod that is installed on the aircraft that senses and transmits aircraft kinematic and response parameters to ground stations. The ground stations monitor, record, and can play back this data for interactive and post fight pilot training.

As this system evolved, it became apparent that the recorded data could provide valuable information for many other applications. One of these applications was to provide aircraft loads parameters during air-to-air tactical training. Table 2 provides a list of some important aircraft loads parameters that are available in time history form. To accomplish this, two sets of computer software were developed. The first was to read TACTS range data tapes to extract and load the needed aircraft and loads data into a computer data bank. The second set of software would select desired data from the computer data bank and carry out loads spectrum analysis functions.

There have been some operational difficulties that have adversely effected the utilization of this TACTS range data as a ready source of combat training loads spectrum data. These difficulties have stemmed primarily from delays in updating the loads data software for consistency with TACTS range software updates. In spite of inoperative periods, this data has contributed to the development of the fatigue test spectrum for the F-4S aircraft, and is currently being used to prepare detail requirements for the VTX and the A-6F aircraft.

The TACTS range data bank will continue to provide an up-to-date source of combat training loads spectrum data for new aircraft development. Since tactical aircraft expend a high percentage of their fatigue life in combat training, this can be a valuable contribution especially if applicable data is not readily available from other sources.

TABLE 2. PARAMETERS USED FOR LOADS SPECTRUM ANALYSIS

o Weight (derived)	o Roll Rate
o Mach Number	o Pitch Rate
o Calibrated Airspeed	o Yaw Rate
o Normal Acceleration	o Roll Acceleration (derived)
o Side Acceleration	o Pitch Acceleration (derived)
o Axial Acceleration	o Yaw Acceleration (derived)
o Roll Angle	o Angle of Attack
o Pitch Angle	o Angle of Sideslip
o Yaw Angle	o Altitude

70 MM FILM LANDING PARAMETER MEASUREMENT SYSTEM

The Naval Air Development Center uses a 70 mm film photographic system for recording landings to determine approach and touchdown parameters. The system consists of a 70 mm half-frame camera to photograph aircraft landings, film readers to provide a means of digitizing flight path motion of the approaching aircraft on a frame by frame basis, and computer software and analysis techniques that determine the desired approach and touchdown parameters for each landing. A second computer analysis routine provides indepth statistical analysis of the individual landing parameters to provide survey results for groups of aircraft generally broken down by type, model, and series. The hardware items are shown in Figure 9. Figure 10 shows a typical setup for a carrier survey. The setup dimensions are input to the analysis program to enable the application of the system to different size carriers and also for field surveys. The film readers project the image on a screen for the operator to index key points on the aircraft such as wing tips, control surfaces, and wheel positions. The readers are tied electronically into keypunch machines that digitize the coordinates of the keyed points. The analysis programs read the card input data and calculate approach and touchdown kinematic and attitude parameters for each landing. Some of the most used parameters provided by this system are listed in Table 3.

The results from these surveys are used to adjust wind over deck requirements, adjust or impose limits on aircraft landing weight, and are the only source of landing data available for development of new aircraft detail design requirements. Some additional tasks that this system has been used for are:

a. Helicopter hovering stability tests.

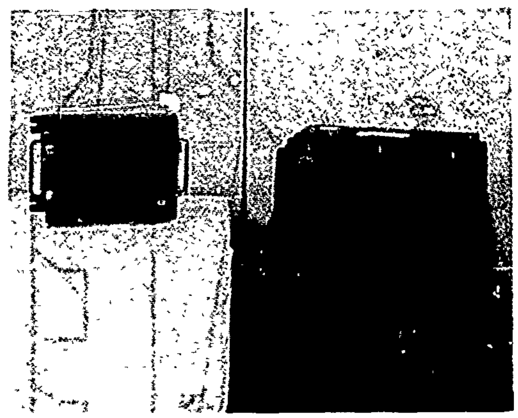
b. Escape system acceleration and trajectory testing.

c. Helicopter landings on small ships.

TABLE 3. PARAMETERS MEASURED WITH LANDING LOADS DATA ACQUISITION SYSTEM

- o Sinking Speeds
- o Approach and Engaging Speeds
- o Minimum Useable Airspeed for Jet Aircraft
- o Lift Factor
- o Aircraft Pitch and Roll Angles
- o Aircraft Pitch and Roll R tes
- o Distance from Ramp to Touchdown
- o Off Center Distance
- o Wheel and Hook Heights Above Ramp
- o Glide Path Angle
- o Carrier Deck Pitch and Roll Angles

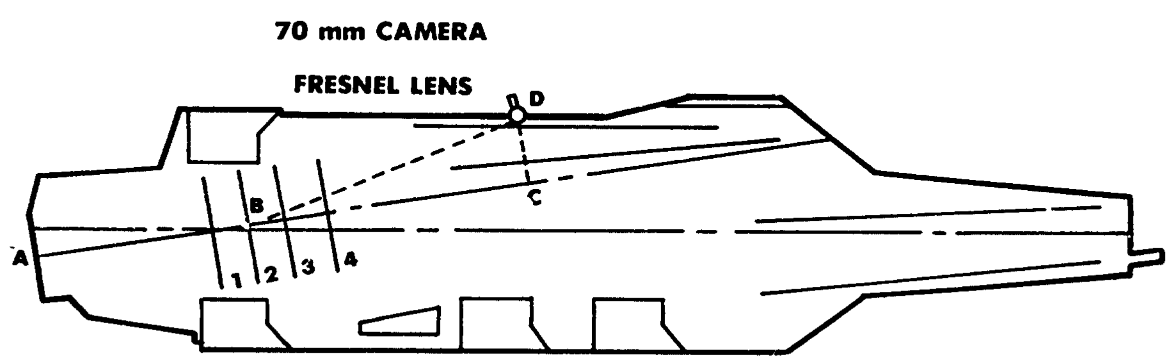
70 MM PHOTO SYSTEM



INFRARED FLASHER

FIGURE 9. 70 MM Film Landing Parameter Measurement System

DECK LAYOUT – USS CARL VINSON – (CVN-70)



RAMP TO WIRE DISTANCE – FEET				CAMERA AND LENS DIMENSIONS – FEET		
1	2	3	4	AB	BC	CD
177.1	217.1	257.5	298.3	298.5	278.5	72.6

Figure 10. LANDING PARAMETERS MEASUREMENT SYSTEM

DISCUSSION OF SERVICE LIFE ASSESSMENT PROGRAM (SLAP)

To provide a more detailed insight into the SLAP Program activity some of the past and current efforts and results will be described.

As mentioned previously, the SLAP Programs consist of analysis, full scale fatigue tests, and Flight and Ground Loads Surveys. The results of Flight and Ground Loads Surveys are generally the backbone of the other efforts. These Survey results combined with the individual aircraft tracking (e.g. SAFE) program results will identify the need and provide the basis for additional analysis and test programs. The Landing Loads Survey results are used to adjust wind over deck requirements for carrier landings and provide the only source of landing data for new aircraft detail design requirements. The Flight Loads Surveys play a major role in determining aircraft longevity and modification or extension program requirements.

The 70 mm film landing loads data measurement system has been used since 1964 to conduct 31 Landing Loads Surveys. Each survey normally consists of filming from 500 to 1,000 landings and are generally conducted on-board aircraft carriers during predeployment pilot carrier qualifications. Field landing and deployment surveys are also accomplished, but less frequently.

A recent Field Landing Survey conducted at a Naval Air Station during a period of routine flight operations revealed that carrier aircraft seldom experience the low sink rate "Field Landing" that is defined in the design specifications. These aircraft normally land at fields like they would if they were conducting Field Carrier Landing Practice.

The most recent carrier Landing Loads Surveys have shown that several of the aircraft models land at higher approach speeds with higher sink rates than expected. They have also shown that for some models the approach speed and sink rates increase as the aircraft get older. The F-4 is an example of this: the mean sink and approach speed values increased from 12.6 ft/sec and 145 knots, respectively to 15.1 ft/sec and 155 knots between 1974 and 1980.

One of the pressing problems that effects most of the current Fleet aircraft relates to the validity of the fatigue tests that were conducted in years past using blocked load spectrums. The U.S. Navy relies heavily on the results of these tests for structural integrity substantiation, and to provide a basis (fatigue allowables) for individual aircraft tracking programs. In order to properly account for sequential effects, these tests must now be redone and all new tests must be conducted with representative flight by flight test load spectrums.

The sequential loads interaction effects are particularly significant on Navy carrier aircraft because of the single flight stress reversals that are experienced in catapult and arresting load sensitive structure. Testing using representative flight by flight spectrums have shown lives reduced by a significant factor from the blocked spectrum results. Figure 11 illustrates the load cycle difference between blocked and flight by flight loads for the A-6 aircraft catapult load reactive structure. The status of the flight by flight load spectrum tests are:

Completed	Underway	Planned
S-3	E-2C	A-6E
F-4S	EA-6B	KA-6D
E/TA-3	F-18	KA-3B
	F-14	
	A-4	

The development of the E-2C fatigue test spectrum is an excellent example of the application of flight and ground load survey data. Since the E-2C was not procured with a fatigue life tracking system, the fatigue life status was initially determined by mission analysis with flight severity extrapolated from its sister aircraft (C-2) data. It was recognized that this technique only provided a barometer for measuring fatigue damage which became even less credible as the aircraft mission and configurations changed with time. The initial concerns centered around a suspicion that these aircraft might be experiencing a high frequency of load factors just below the 2 G level which was the low threshold for the recorded C-2 flight data. To evaluate this, and establish a load factor (N_z) spectrum for the new fatigue test, a 16-airplane, 6-level counting accelerometer load survey was initiated.

The results from this survey were startling. The measured data showed substantially higher than expected frequencies and severity for N_z exceedance levels. Figure 12 shows the proposed spectrum that was initially provided for the fatigue test and the spectrum that was used after the Flight Loads Survey.

This example highlights a couple important issues. The first is the need for accurate loads monitoring systems to preclude making errors in judgment regarding usage severity. The second is that a relatively small surveillance program, such as this one, can really make a difference and is worthwhile, even if all it does in some cases is to establish confidence in major program results.

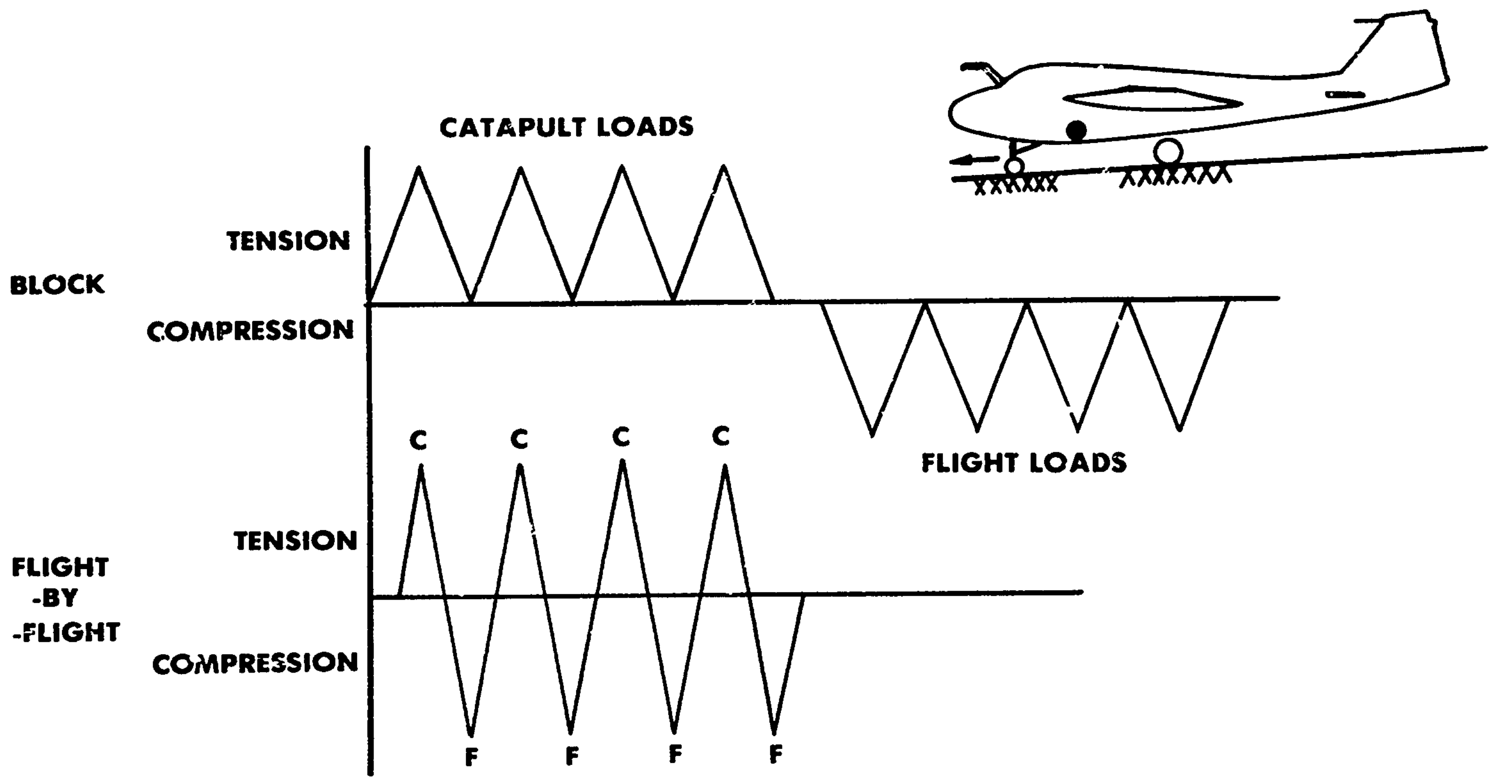


Figure 11. FLIGHT-BY-FLIGHT SPECTRUM

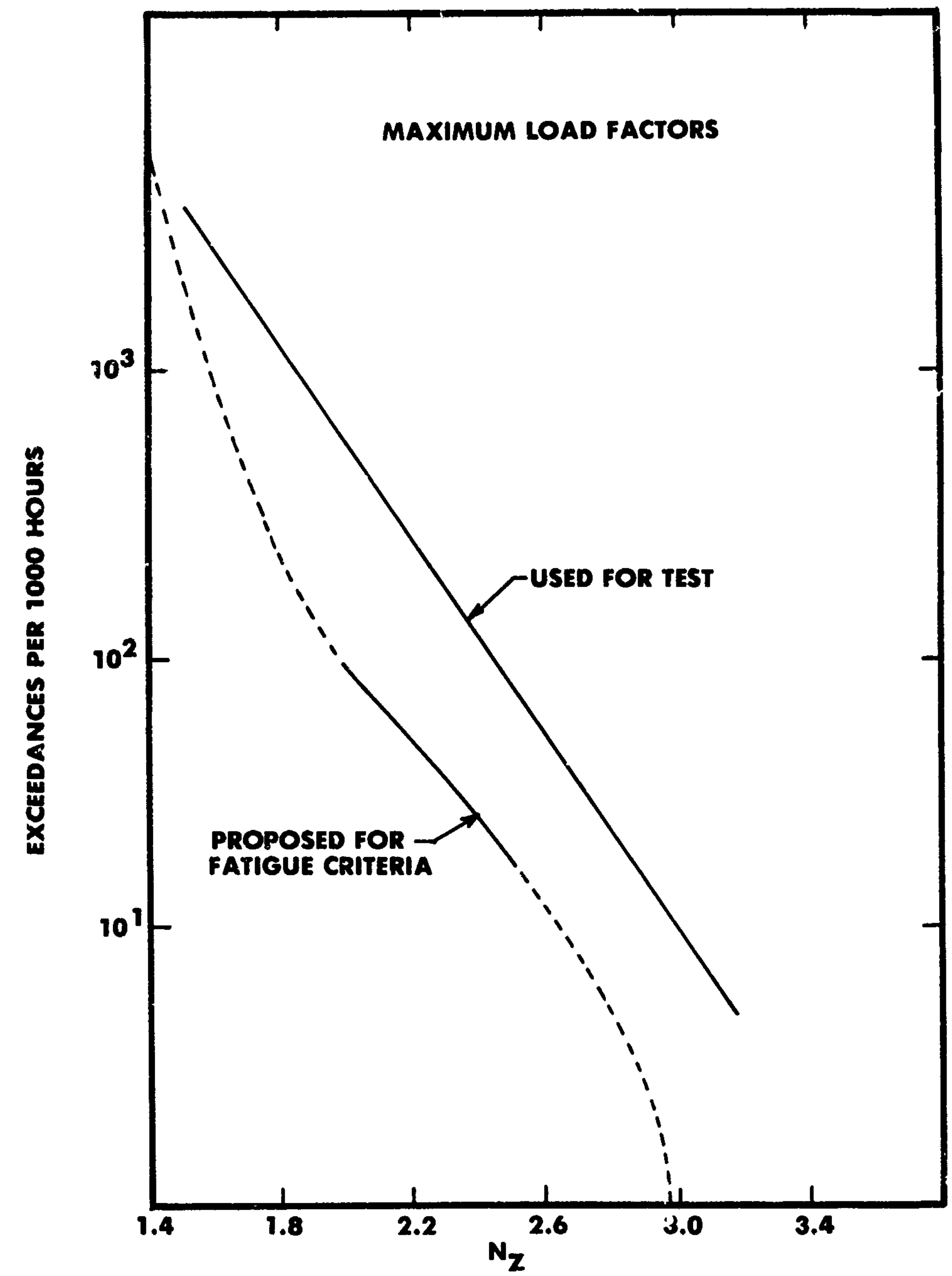


Figure 12. COMPARISON OF EXCEEDANCES OF N_Z PER 1000 FLIGHT HOURS

Other flight usage survey work that has been recently completed or is planned for the near term includes:

Recent	Near Term
F-14 FFLS	A-6E
CH-53D	EA-6B
EC-130Q	P-3C
AV-8C	CH-53E
E-2C	AH-1T
RH-53D	TAV-8A

SERVICE LIFE EXTENSION PROGRAM (SLEP)

SLEP Programs include the engineering, test, and aircraft modification efforts needed to extend the life of aircraft beyond currently authorized limits so they can continue to perform their mission until it is no longer needed. This has become commonplace with Navy aircraft as the original design life of current Fleet aircraft generally falls short of their operational usefulness.

These programs sometimes require incorporation of improved fatigue life monitoring systems (e.g. A-3). These requirements are evaluated on a case by case basis and are prescribed to address the specific aircraft concerns and needs. For instance, the A-3 will get a new Fatigue Life Tracking Program whereas continuation of the current program is planned for the A-4.

The results of the individual aircraft tracking (e.g. SAFE) and Flight and Ground Load Survey Programs play a primary role in determining what needs to be done to make these extensions possible, and in establishing the scope and scheduling for incorporating modifications.

CONCLUSION

The major elements of the Aircraft Structural Life Surveillance (ASLS) Program provides an effective and economical means of assuring safety and longevity of Fleet aircraft airframes. With the advent of multiparameter recording equipment, the effectiveness of the ASLS Program will be even further enhanced. Some of the cost and emphasis will probably shift from the Service Life Assessment (SLAP) Program to the Structural Appraisal of Fatigue Effects (SAFE) Program as more of the multiparameter tracking programs become operational. Though extensive survey work should diminish as more Fleet aircraft are equipped with multiparameter tracking systems, unanticipated problems will always require the measurement of additional data. For this reason, the U.S. Navy plans to employ the new microprocessor technology to maintain and streamline Loads Survey capability. Particular emphasis will be placed on improved systems to measure approach and landing parameters and for direct measurement of landing gear loads. The streamlining efforts will focus on the system packaging, aircraft interface and installation requirements and procedures, data collection and processing requirements and procedures, and analysis and reporting techniques. These efforts will focus on increasing the quality and quantity of data while reducing the costs and response time that are associated with the current Loads Survey aircraft instrumentation equipment and techniques.

EDITOR'S NOTE

The above paper was not presented during the Specialists' Meeting as planned, due to the unavoidable absence of both authors. After giving the matter careful consideration after the Meeting, members of the Panel gave special dispensation for this paper to be included in its proper place in the published proceedings, for completeness. However, preprints were not available at the meeting and there was no opporunity for discussion of the paper.

OPERATIONAL LOADS DATA EVALUATION FOR INDIVIDUAL AIRCRAFT FATIGUE MONITORING

by

Dipl.-Ing. Richard Schütz

Dipl.-Ing. Rolf Neunaber

Industrieanlagen-Betriebsgesellschaft mbH
8012 Ottobrunn, Einsteinstraße, Germany

SUMMARY

The individual aircraft tracking (IAT) can be an effective instrument for introducing the necessary maintenance activities which need to be adapted individually to each aircraft.

One of the most important activities within the IAT scope is the processing of operational loads data in order to calculate to consumed fatigue life of individual aircraft. The installed operational service load recording systems for military aircraft within the German Air Force include logforms, counting accelerometers and digital flight recorders. Under consideration are systems for direct measurement of loads and damage.

Hence, a profound operational loads data acquisition and evaluation can be an effective tool for cost savings during the operational life time of an aircraft.

1. INTRODUCTION

Aircraft structures are subjected to growing demands with respect to complexity of missions and configurations (multirole). For that reason there is a tendency that the complexity of modern aircraft structures is also increasing as it can be seen in the variable sweep wing technique for example.

For aircraft already in service, the fatigue life needs to be reviewed and proved again by experimental and/or analytical studies.

Individual aircraft tracking (IAT) considers as the most effective way to determine the necessary maintenance actions for complex structures.

One of the most important activities within the IAT scope is the processing of operational loads data in order to calculate the consumed fatigue life of individual aircraft. The installed operational service load recording systems for military aircraft within the German Air Force include logforms, counting accelerometers and digital flight recorders. Under consideration are systems for direct measurement of loads and fatigue life consumption.

Hence, a profound and comprehensive operational loads data acquisition and evaluation will be an effective tool for cost savings during the operational life time of an aircraft.

2. OBJECT OF FATIGUE MONITORING PROGRAMS

In service, generally each individual aircraft is subjected to different operational loadings, which will cause different damages in its fatigue prone areas of the airframe structure. FIG 1 shows these facts exemplified on the wing lower skin of a fighter aircraft.

For aircraft having flown the same number of flight hours in comparable missions (fighter bomber) a considerable scatter in consumed fatigue damage up to a factor of 5 could be observed. Consequently, a factor of 5 exists between the mildest and the most severe flown aircraft considering identical flight hours. If no fatigue related monitoring programme is carried out, maintenance actions like inspections, modifications, exchange or scrap of components have to be done at the number of flight hours which the most severe flown aircraft is allowed to accumulate taking into account a defined probability of failure. That means, that for all remaining aircraft, the relevant maintenance actions come too early since they are flown in a milder way.

The fatigue monitoring program allows the maintenance actions to be introduced on the basis of the accumulated damage rate which can be calculated for each aircraft if loading is recorded for each individual aircraft. Therefore maintenance actions for a fatigue critical area will not primary be defined by accumulation of a certain number of flight hours, but after reaching a damage rate of 100 %, which will mean a different number of flight hours for each aircraft.

With consideration of the life already consumed and under certain assumptions for future operation the remaining flight hours for utilization before maintenance actions fall due, can be determined. Additional logistic data concerning required tools, spares and personnel can be defined more precisely with the aid of fatigue monitoring.

With mission specific load sequences, extracted from recorded data of individual aircrafts an operation control is possible to a certain extent. So, for instance aircrafts with high damage rates could allocated to fly less severe missions and/or configurations, whereas aircrafts with low damage rates can be rotated to high damaging missions.

3. INDIVIDUAL AIRCRAFT TRACKING PROGRAM (IATP)

The purpose of the IATP is to determine the attained damage as percentage of the allowable fatigue life or, with other words, to determine the consumed life of each individual fatigue critical structural component. FIG 2 shows the main activities of the IATP:

- LOADS AND COMPONENT DATA ACQUISITION

  With individually adjusted load monitoring systems the relevant load parameters will be recorded for all flights for each aircraft. Additionally the component identification information (serial number) is registered.

  These data are forwarded to the Central Evaluation Department (CED).

  Special checks are applied in order to separate, correct or replace faulty data.

- DAMAGE CALCULATION

According to the design philosophy of the aircraft structure, the appropriate damage calculation philosophy is applied:

- o for safe life structures the calculation is based on S/N curves and the Miners rule [1], [2] is used to determine the accumulated damage. Usually the loads data will be processed to a cumulative frequency stress spectrum.

- o In damage tolerant designed structures initial flaws are assumed in the new built structure. They are monitored until defined crack sizes are reached, depending on the associated maintenance action (economic repair, replacement, inspection). In order to calculate the crack propagation, the cycle by cycle information of the stress sequence is considered to obtain special retardation effects [3], [4].

- DERIVATION OF STANDARD LOAD SEQUENCES OR SPECTRA (SLS)

Out of the registered loads, typical sequences are extracted in order to create specific parameter or load histories for full scale or component tests as well as for reference spectra (standard unit spectra, SAS). These load sequences or spectra should satisfy the following criteria:

- o the mean damages of the registered load sequences of individual aircraft should be equal to the mean damage of the SLS

- o the distribution of actual flown missions, configurations and other relevant operational parameters (e.g. vertical acceleration, flight duration, altitude, velocity) should be characteristic for the aircrafts operational usage. In some cases more than one SLS or spectrum has to be determined for the weapon system under consideration e.g. different SLS for combat and training missions.

- FATIGUE LIFE SUBSTANTIATION

The fatigue life of aircraft structural components is demonstrated on the basis of relevant requirements [5], [6], [7]:

- o Full Scale Tests
  Major structural components such as center fuselage, wings and empennage are fatigue tested in complex rigs, the fatigue critical areas found are investigated with special fracture mechanic related methods.

- o Component Tests
  For special substructures like landing gear, stabilizers, forward fuselage portions, separate tests are carried out.

- o Arithmetical Analysis
  In case of structural modifications due to insufficient fatigue life or for minor subassemblies theoretical analyses have to be done.

- o Operational Experience
  Additional operational experience of each in-service loading can be helpful to correct test and analytical results.

The loading program for the fatigue tests and the arithmetical analyses should be based on the SLS.

If the relevant tests are carried out in the structural design phase, estimated load sequences which are often derived from similar type of aircrafts must be used. For the operational usage phase the fatigue test life is scaled to a SLS related fatigue life.

- CALCULATION OF ALLOWABLE FATIGUE LIFE

The allowable fatigue life depends on the planned maintenance action

- o inspection
- o economical repair
- o replacement/scrap

and the design philosophy

- o safe life
- o damage tolerance

For safe life designed structures the scatter of fatigue life is related to a low probability of occurance of a damage which in turn is dependend on the structural safety class of the considered component (FIG 14).

For damage tolerant structures the flaws assumed to exist in the new built structures and which escaped detection during quality control are related likewise to a low probability of occurance.

The attained life at the crack size, which is associated to the planned maintenance action, is the allowable life for a damage tolerance structure.

The allowable life is always related to the standard load sequence or spectrum.

- CONSUMED LIFE, DAMAGE RATE

The damage of each component in each individual aircraft is related to the allowable life. The ratio in percent represents the consumed life or damage rate.

FIG 3 shows a typical printout of the individual aircraft tracking program for a fighter bomber. It should be noted that in case of symmetrically arranged components, like wings, it is attempted to combine components with nearly equal damage rates in order to synchronize the inspection activities.

- PREDICTION OF MAINTENANCE ACTIONS

In order to get information for planning inspections (personnel and devices), exchange (spare part availability and provisions) or economic repair (tooling) a prediction method is applied which contains the following criteria

- o the aircraft is flying in the same squadron in future
- o the mean squadron load sequence or spectrum is used for the prediction
- o the maintenance action to fall due is integrated into the next suitable depot or field maintenance

In FIG 3 the predicted flight hours until a maintenance action must be taken (100 %) is listed for different fatigue critical locations.

4. LOAD MONITORING SYSTEMS (LMS)

The primary object of load acquisition is to get information from the fatigue critical areas in terms of strain or stress to calculate the damage in incremental steps per load cycle. The systems which are used or which are under consideration today for IAT tasks can be divided into

- flight parameter measuring systems
- direct strain measuring systems
- on board damage calculation systems
- load cumulation systems

The selection of the most suitable system for a defined fleet of aircraft depends on the following criteria (FIG 4)

- COMPLEXITY OF THE STRUCTURAL DESIGN AND LOCATION OF FATIGUE CRITICAL AREAS
  The higher the degree of complexity of the structural design, the more complex is the data acquisition system, e.g. for fixed wing combat aircraft designed in the mid 50's counting accelerometer, partly in combination with simple recorders (VGH), are sufficient for IAT purposes. For highly complex structures of sweep wing aircraft highly sophisticated data acquisition systems are necessary.

  Fatigue critical areas located at inner wing/centre fuselage normally can be monitored by counting accelerometer systems other locations need further systems.

- COSTS EFFECTIVENESS CONSIDERATION
  The total costs - the sum of system hardware and regular evaluation - for the IAT should be a minimum, since the amount of data from IAT can be tremendous.

  Therefore, data acquisition systems should have a high reliability level, otherwise, due to system failures additional repair costs as well as penalties for the consumed life must be taken into account.

  A concept with a simple data acquisition in all aircrafts and a complex system in a few aircrafts has been proved as good compromise.

  Within the scope of a basic investigation the total fixed and current costs of the applicable load monitoring systems are compared with the expenditure of costs to carry-out the necessary maintenance actions. Generally, the maintenance action costs, economized by load monitoring, must be considerably higher than the fixed and current fatigue load monitoring expenses.

- REQUIRED DATA DISINTEGRATION
  Repeated loads result in damage accumulation for safe life structures or in crack propagation for damage tolerant structures. For both damage calculation philosophies the strain or stress conditions at each load maximum and load minimum have to be determined at fatigue critical locations. Since at these locations the elastic and plastic loading reactions can not be described exactly, considerations concerning a concept to overcome such deficiencies have to be investigated.

  A suitable transferfunction T is to be defined in order to describe the correlation between the relevant strain/stress at the starting point of the crack and the registered IAT parameters or recorded strain/stress sequences [8].

4.1 FLIGHT PARAMETER MONITORING SYSTEMS

Flight parameter monitoring systems are applied primarily to data acquisition systems installed in all aircraft as well as to sophisticated data acquisition systems installed in a representative sample of aircraft. One of the most widespread acquisition systems for the vertical acceleration of aircraft is the

- COUNTING ACCELEROMETER (g-METER) (CA)
  This device is a relatively simple indicator with a relatively high degree of reliability. If the variation of fatigue relevant ` eters other than vertical acceleration is negligible, the reading intervals can be chosen from 25 to 100 flight hours, otherwise a flight by flight reading with additional log-recorded parameters is necessary, especially in the case of highly variable external store configurations. The relevant transfer functions T1 and T2 (FIG 5) are applicable. These transfer functions contain a large portion of estimated data which are based on statistic.. assumptions, so that a defined worst case is covered. Since a specific aircraft is not always flown under severe loading sequences and another one not always under exclusively mild loading sequences but rather a mixture of both loading types, a special scatter consideration is necessary. In FIG 6 these facts are shown for typical fighter bomber squadrons with the characteristic decrease of damage scatter for increasing number of flight hours. Therefore, if the lifetime of components is long enough, the statistically based load data are in an acceptable range. The structural areas which can be monitored by counting accelerometer are essentially limited to

  - center fuselage
  - inner wing

  Since the hardware costs of counting accelerometers are relatively low, this system is used mostly in 100 % of the aircraft of a certain fleet.

- DIGITAL FLIGHT PARAMETER (DFP) RECORDER
  The function of the DFP-Recorder is to register, to digitalize and to store the values of defined parameters during a flight onto tape. By means of a transcriber the data are converted into a format compatible to processing. Two categories of DFP-Recorder which differ mainly in data volume, can be discerned:

  <u>Velocity-Gravity-Height (VGH) Recorder</u>
  For aircraft structures with a relatively low complexity a VGH recorder in combination with a flight log, containing information on weight at start and landing as well as configuration is useful. With a transfer function of T3 (FIG 6) type in which unsymmetrical loads must be estimated on a statistical basis and special load effects - e.g. buffet, tip stall - are integrated, the registered data are transduced to a peak and through stress sequence for damage calculation. Since the expense in calculation is in most cases tremendous, the VGH recorder is used as an additional data acquisition system in about 15 to 20 % of all aircrafts in combination with CA in 100 % of the aircrafts.

  Due to the electronic micro-miniaturization the potentiality of VGH-type recorders will increase so, that at the same hardware-weight a multichannel recorder or an on board damage calculation system will be available.

The major disadvantage of the VGH-recorder against the multichannel recorder is the missing information on weight conditions, configuration and unsymmetrical loads. These informations must partly be handwritten on a special log form. Essentially the

- front and center fuselage
- inner and outer wing structures

are covered with the VGH recorder. The percentage of data drop outs for the whole system, consisting of the recorder unit and the transcriber, comes to less than 5 %. A typical sequence of vertical accelerations with a specific data trouble (spikes) and their correction demonstrates FIG 7.

o MULTICHANNEL (MC) RECORDER

In principle the MC recorder is an extended VGH recorder, registering additional sequences of aircraft parameter such as angular rates and lateral accelerations, fuel quantities, external store drops. FIG 8 shows the structural relevant parameters with the respective sampling rates for a planned MC recorder, at the moment under development for a complex sweep wing fighter aircraft.

As it can be seen, provisions are assigned to operate strain gauges. With these strain gauges a double evaluation is possible:

- for fatigue critical locations, which are covered by the transfer function type T4, a regular correlation- and regression analysis between the flight parameter and the strain or stress at the fatigue critical area and thereby an actual adaptation of the transfer function T4 can be performed

- for fatigue critical areas, which cannot be covered by registered flight parameters such as landing gear and empennage, from the direct strain measurements representative stress/strain sequences or spectra can be derived.

With this ability the MC recorder comes into the category of direct strain measuring systems. Likewise strain gauges are helpful if an unexpected critical location occurs during service and a quick estimation of the possible fatigue life is necessary.

Because of the comparatively high datavolume an installation into 10 to 20 percent of aircrafts in the fleet is reasonable. Data loss due to sensor defects or electromagnetic disturbances add up to 15 to 20 percent.

The registered VGH and MCR recorder data must be evaluated so, that a level-correlated damage-transfer on a mission specific-basis gives the possibility to calculate damage relevant indices from counting accelerometer readings.

4.2 DIRECT STRAIN MONITORING SYSTEMS

The fundamental parameter for any damage calculation procedure is the stress/strain history at the location, where a crack occurs and propagates with load assoziated rates.

Besides the possibility to integrate strain gauges into the MC recorder, an important system for strain registering is the

- MECHANICAL STRAIN RECORDER (MSR)

  The MSR works as a mechanical strain gauge, "scratching" the sustained strain history onto a metallic tape stored in a cassette (FIG 9). One scriber scratches the record trace, another scriber is scratching a reference trace, whereby the difference between these two traces is a measure for the applied strain [9].

  Due to a backlash of the recorder and a minimum peak solution of the data transcriber unit, amplitudes greater than 0.0025" are registered.

  The available gauge lengths measure 3, 5, 8 and 10 inches. The 8 inch MSR proves to be an acceptable compromise for aluminium structures of transport aircraft, since the strain solution is sufficient, an adequate calibration can be carried out.

  The MSR is at the moment installed for test purposes in 4 transport aircraft wings in the GAF (FIG 10). This component offers enough room for bonding a MSR with good accessability and the most fatigue critical location of the wing is near a spliceplate for which a transfer function T4 (FIG 5) can be calculated. So far the measured time with MSR installed totals up to approximately 500 flight hours. Within this period an interval of about 35 flight hours was not reproducable, because one mounting block had loosened.

  The necessary transfer function T5 was determined by conventional strain gauge calibrations. The calibration after each cassette removal is done by filling the fuel tank with a defined amount of fuel, producing a difference in wing bending moment.

  The results of the test operation usage show that an extension of the MSR installation into all transport aircrafts can be effective, especially since no additional data needs to be read by the crew as it is necessary up to now with other systems.

4.3 ON BOARD DAMAGE CALCULATION SYSTEM (OBDCS)

A "weak" point of the DFP Recorder (VGH and MC recorder) is the data transfer procedure, that means writing the digital information onto a magnetic tape in the DFP recorder, subsequently transferring the DFP recorder data via transcription unit onto a computer compatible magnetic tape and format for damage calculation. The loss of data reaches about 5 to 20 percent depending on the adjustment of the micros in the recorder and the data transcription unit.

Therefore, planning studies are initiated, to calculate the damage in an OBDCS. The subprograms necessary to calculate the damage according to a linear damage accumulate method are shown in FIG 11. For each subprogram the estimated storage capacity for random access memory (RAM) and read only memory (ROM) as well as erasable programable read only memory (EPROM) are shown in this figure.

The dominant subprograms contain the following procedures

- DATA CONTROL AND CONDENSATION
  In this routine the incoming data are checked with respect to gradient and amplitude and, if necessary, corrected. In a second step the values are condensed to an adapted peak and through sequence, considering a threshold for elimination of noise from the used record.

- STRESS CALCULATION FOR EACH CRITICAL LOCATION
  With application of the function of T4 type (FIG 5) bending moments and/or shear/ tension forces are determined out of the recorded flight parameter and then the stress is calculated for each critical location by a linear function.

- RANGE PAIR COUNTING
  In this subprogram the number of stress ranges are calculated, taking into account the mean level and a specified threshold. In addition the rain flow cycle counting method is applied in order to fix the damage relevant cycles and half cycles [10].

- DAMAGE CALCULATION
  The damage calculation is based on a conventional damage accumulation method according to Palmgren/Miner. For each fatigue critical location a set of S-N data is stored in an EEPROM for easy exchange of data. The damage calculation is carried out on demand only.

- RESULT STORAGE
  The actual damage sum for each fatigue critical location is stored in an NV RAM, the values can be read out by a hand held terminal on demand. The individually accumulated damage rate of a component is calculated as the quotient of the component damage sum to the allowable damage sum.

Beside the above described function, the OBDCS is used for additional tasks, for instance to control the flight limitations and other defined events.

4.4 LOAD CUMULATION SYSTEMS (LCS)

At the moment two systems are taken into consideration:

- FATIGUE GAUGE (FG)
  The FG resembles in appearance a foil strain gauge. When bonded near a fatigue critical location, the FG creates an electrical resistance change as a function of the applied load [11].

  For evaluation of the endured fatigue damage, two different types of FG are necessary a directly bonded FG and a multiplied FG.

  Fatigue critical locations develope in most cases at fastener holes with no direct accessability. Fatigue gauges are therefore in these cases not applicable, since an direct extrapolation from a location of varying stresses is impossible.

Another problem arises with the requirements concerning the Ohmmeter in service usage. The measured resistance values between two reading intervals can be in the range of the measuring accuracy. Therefore, FG's are used until now only in laboratory applications.

- EMS (SYSTEM FOR MONITORING FATIGUE LOADS)
  The EMS, under development by DORNIER is based on specially prepared, polished and annealed metal foils, which are bonded near the fatigue critical locations of components.

  A restriction is, that the location must be accessible for an optical reflectance measuring (ORM) system [12].

  FIG 12 describes the EMS evaluation steps:

  o EMS ON FULL SCALE FATIGUE TESTING
    For full scale fatigue testing to each possible fatigue critical location an EMS is bonded. At defined intervals of test hours the momentary degree of reflectivity is monitored with an ORM System and related to the initial value (relative reflectivity). The allowable life is defined and the relative reflectivity is registered.

  o SPECTRUM VARIATION
    Tests with special specimens are carried out whose stress distribution equals as far as possible that of the FSFT with a variation in frequency and amplitude of the stress at a defined fatigue critical loaction. Simulation of the variation in service gives the basis for the fatigue monitoring program.

  o FATIGUE MONITORING WITH EMS
    Each individual aircraft of a certain fleet will be supplied with one or, for redundance, two EMS at each fatigue critical location found in the tests.

    At defined intervals, about once a month, the EMS reflectivity is registered with the ORM system and sent to the CED. The reported reflectivity values are interpolated to the "life line" from which the consumed life and the remaining life will be calculated.

  At the moment two foil materials are under evaluation: aluminium and tin. Tin has, compared with aluminium, a higher sensitivity but likewise a higher failure rate. This problem has to be solved before operational usage comes into question.

5. VERIFICATION OF FATIGUE LOAD MONITORING

The validity of recorded loads can be confirmed by investigation of cracks occuring during operational usage. A striation counting method was developed in order to correlate the calculated or measured stress cycles to the crack propagation investigated on the failed component [13].

Exemplified on a crack which occurred on the lower skin of a fighter bomber aircraft, this method will be described (Fig. 13):

o The cracked bolt hole was cut out from the damaged component and subsequently opened

- Under the electronic scanning microscope the striations were counted as a function of the crack length and integrated over the full crack length.

- The individual load sequence, in this case counting accelerometer readings was randomized to a vertical acceleration sequence and was treated with the transfer function to get the stress sequence.

- A crack propagation calculation with variable parameter along with a linear damage calculation according to Palmgren-Miner's theory was carried out and correlated with the striation evaluation. As a result of this analysis different measures have to be taken:

  - Adaption of the fatigue allowables in form of damage sums

  - Correction of the crack propagation calculation parameters, expecially the retardation behavior parameters

  - Revision of the transfer functions in order to improve the assumptions concerning the estimated parameters

With the principles of fracture mechanics the applied fatigue load monitoring procedures can be verified by analysis of service cracks, if these are utilizable.

6. BENEFITS OF LOAD MONITORING FOR OPERATIONAL AIRCRAFT USAGE

All optimized fatigue relevant maintenance need to be based on the operational loads occurring in a definite interval.

Taking into account, that the fatigue strength and the initial flaw distribution of a structural component scatters due to manufacturing influences such as drilling holes and milling notches with blunt tools as well as heat treatment processes, a possible "weak" element containing a flaw resulting in a low fatigue strength with a defined low probability has to be considered.

- SAFE LIFE STRUCTURES
  If no individual aircraft tracking program with load monitoring is carried out for safe life designed structures, the interaction of scatter in strength and loading must be considered (FIG 14):

  - For a fatigue critical component the statistical distribution of the test hours related to the reference spectrum at the allowable crack length is calculated. The allowable crack length depends on the associated maintenance action: e.g. economic repair by reaming the hole or exchange of the part, but also from inspection method and degree of inspectability.

    Based on results from a lot of representative tests for the standard deviation of fatigue strength a average value of $S_F = 0.1299$ or a scatter of 1:2.15 is assumed. In order to get the mean value of the total population, a risk factor is introduced, calculated according to $\log R = S_F/\sqrt{n}$ with n being the number of test specimens.

- With fatigue load monitoring generally the scatter for the fatigue strength needs be considered because for each individual aircraft the fatigue relevant loads are known. The probability of occurance of the allowable crack length is a function of the structural safety class and typically ranges from 10^{-2} to 10^{-6} for highly critical parts.

- Without fatigue load monitoring the interaction of fatigue strength and the loading variation has to be considered The standard deviation of loading is normally determined from the measured portion of the 1000 flight hours related damage sums of aircrafts in a defined squadron type. Dependend on the flown sorties, the standard deviation of loading varies for combat aircraft in the range from 0.10 to 0.16, for transport aircraft from 0.11 to 0.15, related to 1000 flight hours damage sum.

- In FIG 15 the gain in operational fatigue life due to load monitoring as a functi . of the loading scatter is shown for discrete probabilities of damage occurance. As it can be seen, fatigue relevant maintenance actions are necessary in average 25 % to 100 % later with IAT, depending on the loading scatter and the probability of damage occurance.

  IAT is particularly effective if the flown missions vary in a large scale. This is the case when secondary roles supplement the primary role, for instance when reconnaissance missions are extended by air to ground or air to air missions.

o DAMAGE TOLERANCE STRUCTURES

For damage tolerant structures, monitored by counting accelerometers and digital recorders, a level related damage index can be determined, so, that the g-level correlated portion of crack propagation gives the allowable number of cycles until failure occurs (FIG 16).

If the fatigue relevant loads are not individually monitored, for instance by counting accelerometers (CA), a damage index related to one flight hour, is calculated, so, that a defined probability of occurance is covered. FIG 6 demonstrates a characteristic example for the flight hour dependent scatter of loading with the damage index for unmetered flight time in relation to load monitored flight time.

7. CONCLUSIONS

Fatigue load monitoring i- an effective tool

- to reduce maintenance expense by individual adap ion of the regis red fatigue relevant loads for each critical location

- to extend fatigue life beyond the planned operational period with determination of the additional structural improvements

- to predict the date of maintenance action in order to provide the necessary personnel, tools, jigs etc.

The most suitable data acquisition system for fatigue load monitoring can be determined by the following criterias (FIG 17):

- location of the fatigue critical area
- hardware- and evaluation costs in relation to the effected maintenance effort

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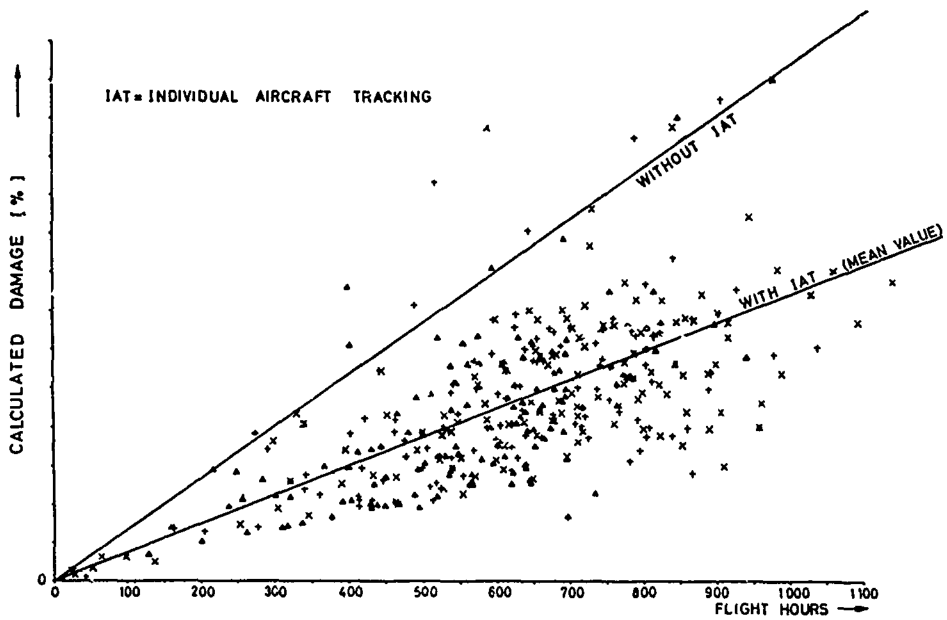


FIG. 1 CALCULATED DAMAGE FOR THE LOWER WING SKIN (FIGHTER BOMBER SQUADRON)

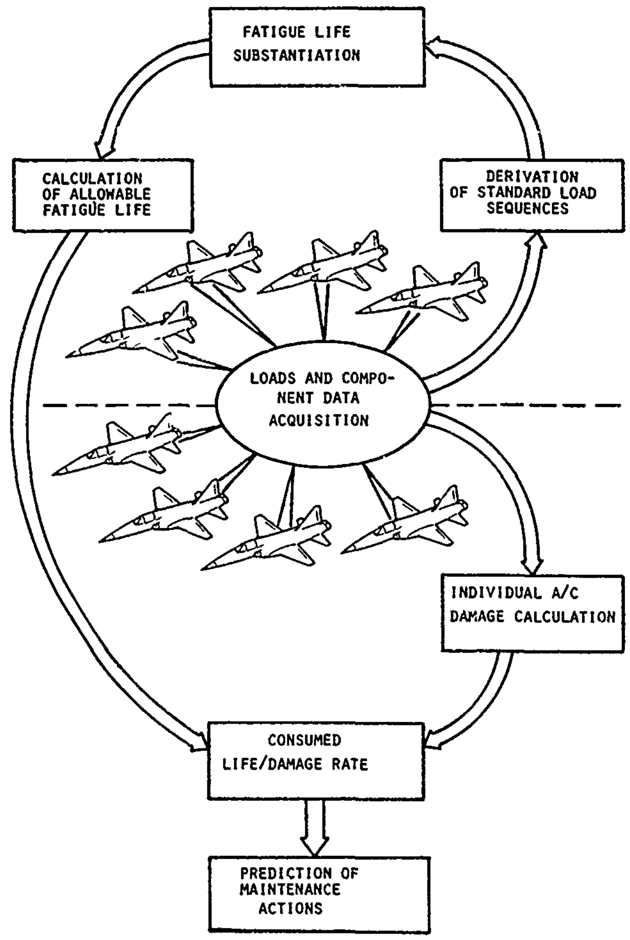


FIG 2 MAIN ELEMENTS OF THE INDIVIDUAL AIRCRAFT TRACKING (IAT) PROGRAM

INDUSTRIEANLAGEN BETRIEBSGESELLSCHAFT ABT. TFA 8012 OTTOBRUNN — DATE

FATI V 01 — DATA UNTIL

L E D A

LIST 1 — WING NO. 01 AIRFORCE BASE — PAGE 1

NO	SERIALNUMBER AIRCRAFT	SERIALNUMBER RH-WI	SERIALNUMBER LH-WI	SERIALNUMBER TRDC.	SERIALNUMBER COUNT.	TOTAL FLIGHT HOURS FUSL.	TOTAL FLIGHT HOURS RH-WI	TOTAL FLIGHT HOURS LH-WI	DAMAGE IN PERCENT FUSL.	DAMAGE IN PERCENT RH-WI	DAMAGE IN PERCENT LH-WI	REMAINING HOURS UNTIL 100CD FUSL	REMAINING HOURS UNTIL 100CD RH-WI	REMAINING HOURS UNTIL 100CD LH-WI	REMAINING HOURS UNTIL INSPECTION RH-WI	REMAINING HOURS UNTIL INSPECTION LH-WI	FIRST WING MONTH/YEAR INSP.	FIRST WING MONTH/YEAR 100CD
1	26	78	14	24	77	2877	913	554	88.34	110.59	63.06	341	0	270	0	0	00/00	00/00
2	50	96	09	79	11	2631	853	853	83.10	106.20	106.20	492	0	0	0	0	00/00	00/00
3	17	36	05	68	22	2483	831	609	76.31	102.40	48.44	494	0	386	0	93	00/00	00/00
4	51	02	61	38	71	2606	702	695	83.60	99.43	89.79	488	4	76	0	0	00/00	4/01
28	80	60	10	.	14	3543	852	852	107.90	57.88	57.88	*0	494	494	33	33	9/81	5/83
29	33	42	30	81	17	2618	434	445	83.34	54.83	55.34	408	338	334	39	38	9/81	10/82
30	07	00	07	95	96	2026	367	499	89.23	53.33	54.01	*317	349	330	50	39	8/81	10/82

S 35 STORED AIRCRAFT
RATD REMOVED

RH-WI: RIGHT HAND WING
LH-WI: LEFT HAND WING

*REMAINING HOURS FUSELAGE LESS THAN REMAINING HOURS FIRST WING
100CD 100 0/0 CALCULATED DAMAGE
00/00 DATE OF INSPECTION OR 100CD IS EXCEEDED

FIG 3 PRINTOUT OF INDIVIDUAL AIRCRAFT TRACKING PROGRAM

NO.	A/C TYPE	PRIMARY MISSIONS	RELATIVE A/C QUANTITY	STRUCTURAL COMPLEXITY	INSTALLED SYSTEM FOR IAT PROGRAM		READING INTERVAL	VARIATION IN EXTERNAL CONFIGURATION	POSITION OF FATIGUE CRITICAL AREAS	TYPE OF TRANSFER. FUNCTION [1]
					PRIMARY (100 % A/C)	ADDITIONAL				
1	COMBAT	• FIGHTER BOMBER • NAVY	HIGH	LOW (FIXED WING)	COUNTING ACCELERO-METER (CA)	FLIGHT LOAD MEASUREMENTS	25 FH	LOW	MID FUSELAGE WINGROOT & INNER WING	T1
2	COMBAT	• GROUND ATTACK	MEDIUM	LOW (FIXED WING)	COUNTING ACCELERO-METER (CA)	FLIGHT LOAD MEASUREMENTS	1 FLIGHT	LOW/MEDIUM	MID FUSELAGE WING ROOT & INNER WING	T2
3	COMBAT	• FIGHTER BOMBER • FIGHTER • RECCE	MEDIUM	MEDIUM (FIXED WING)	COUNTING ACCELERO-METER (CA)	15 % A/C WITH VGH-RECORDER	100 FH	HIGH	FWD & MID FUSELAGE INNER WING OUTER WING	T1 T3
4	COMBAT	• MULTIROLE COMBAT • NAVY	MEDIUM	HIGH (SWEEP WING)	MOMENTARY CA - - - - - PLANNED: ON BOARD DAMAGE-CALCULATION	5-10 % MR [2] - - - - -	1 FLIGHT	HIGH	FUSELAGE WING	T2 T4 - - - - - T4
5	TRANSPORT	• TRANSPORT	LOW	MEDIUM	MOMENTARY FLYING LOG - - - - - PLANNED FOR NEAR FUTURE MSR	- - - - -	MOMENTARY 1 FLIGHT MSR: 100 FH	NO	INNER WING OUTER WING OPTION FOR FUSELAGE	T1 T5

[1] SEE FIG 5 [2] PLANNED FOR NEAR FUTURE

FIG 4 USAGE OF DATA ACQUISITION SYSTEMS FOR IAT

DESIGNATION	TYPE OF FUNCTION	MEASURED PARAMETER BY LMS [2]	ADDITIONAL RECORDED FLIGHT LOG PARAMETER	ESTIMATED DATA ON STATISTICAL BASIS	FUNCTIONAL CORRELATION FOR PILOT LOCATIONS [1]	EXTENSION TO OTHER LOCATIONS i
T1	LINEAR	• VERTICAL ACCELERATION N_Z (ACCUMULATED)	• CONFIGURATION FLIGHT HOURS	• TIME OF LOAD OCCURANCE • FLIGHT DURATION • FLAP POSITIONS IN CONFIGURATION • VELOCITY, ALTITUDE • UNSYMMETRICAL LOADS • CENTER OF GRAVITY • SPECIAL EFFECTS	$\sigma = C \cdot N_Z$	• DAMAGE CORRELATION • $\sigma_i = C_i \cdot N_Z$
T2	LINEAR	• VERTICAL ACCELERATION N_Z (ACCUMULATED)	• CONFIGURATION • FUEL AT START/ LANDING • FLIGHT DURATION	• TIME OF LOAD ACCURANCE • VELOCITY, ALTITUDE • UNSYMMETRICAL LOADS • CENTER OF GRAVITY • SPECIAL EFFECTS	$\sigma = f(N_Z$, WEIGHT, CG POSITION)	• $\sigma_i = f_i(N_Z$, WEIGHT) • DAMAGE CORRELATION
T3	SEGMENTARY LINEAR	• VERTICAL ACCELERATION N_Z • VELOCITY V • ALTITUDE H • FLAP/SLAT POSITION (FSP) • FLIGHT DURATION	• CONFIGURATION • FUEL AT START/ LANDING	• UNSYMMETRICAL LOADS • SPECIAL EFFECTS (BUFFET)	$M_B = f(N_Z, V, H$, FSP, WEIGHT)	$\sigma_i = C_i M_B$
T4	SEGMENTARY LINEAR	• 20 RELEVANT PARAMETER (SEE FIG 8)	-	-	$S, M_B = f$(ALL AVAILABLE RELEVANT PARAMETER)	$\sigma_i = C \cdot M_B, S_i$
T5	LINEAR	N.A.	• CALIBRATION	N.A.	$\sigma = C \cdot \sigma_{MEAS.}$	$\sigma_i = C_i\, \sigma_{MEAS.}$

[1] σ = STRESS M_B = BENDING MOMENT, C = CONSTANT, S = SHEAR, σ_{MEAS} = MEASURED STRESS
[2] LOAD MONITORING SYSTEM

FIG 5 TRANSFER FUNCTION FOR STRESS CALCULATION

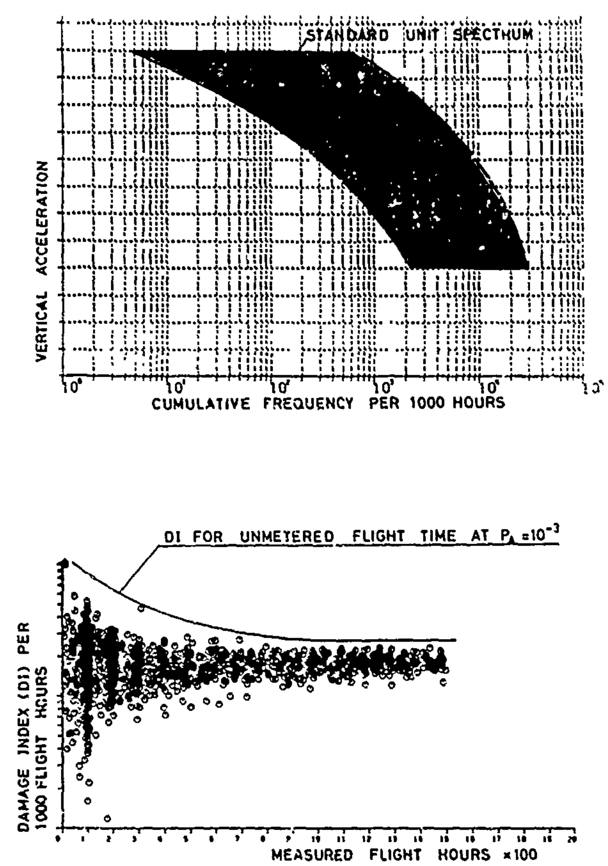


FIG 6 DECREASE OF LOADING SCATTER

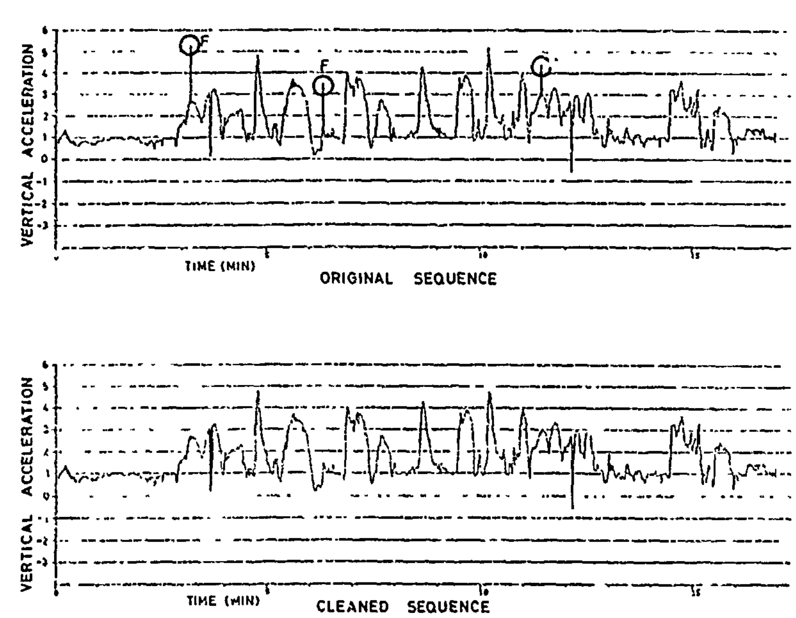


FIG. 7 FAULT DATA DETECTION AND DATA CORRECTION ROUTINE

PARAMETER NO.	PARAMETER	DATA TYPE	SAMPLING RATE
1	PRESSURE ALTITUDE	DS	0.5
2	CALIBRATED AIRSPEED	DS	0.5
3	NORMAL ACCELERATION	A	16
4	TRUE ANGLE OF ATTACK	DS	2
5	ROLL RATE	A	8
6	PITCH RATE	A	4
7	YAW RATE	A	2
8/9	TAILERON POS. (PT/STBD)	A	2x4
10	OUTBOARD SPOILER PT	A	1
11	INBOARD SPOILER STBD	A	1
12	RUDDER POSITION	A	2
13	WING SWEEP ANGLE	A	0.5
14	ELAPSED TIME	INT.	0.5
15/16	SPARE 1/2 (STRAIN GAUGE)	A	16/4
17/18	FUEL FLOW (PT/STBD)	F	1/1
19	FLAP POSITION	A	1
20	SLAT POSITION	A	1
21	INITIAL DATA & SPARE	DS	1
22	INITIAL DATA & SMS	DS	1
23	INITIAL DATA, SPEC. WEAPON	DS	1
24	OLEO SWITCH	D	0.5

A = ANALOG
F = FREQUENCY

FIG. 8 STRUCTURAL RELEVANT PARAMETERS FOR A MULTICHANNEL RECORDER

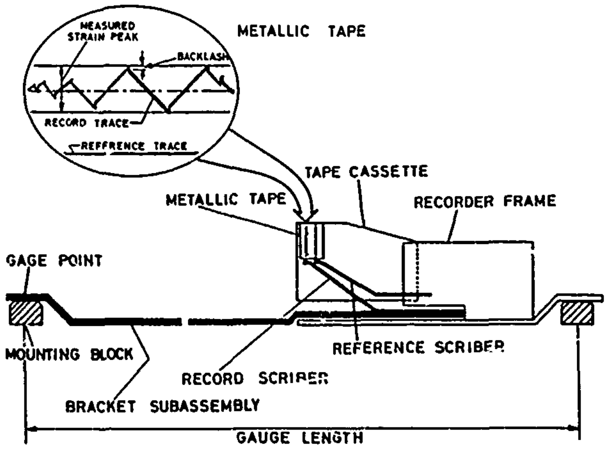


FIG. 9 PRINCIPLE OF MECHANICAL STRAIN RECORDER (MSR)

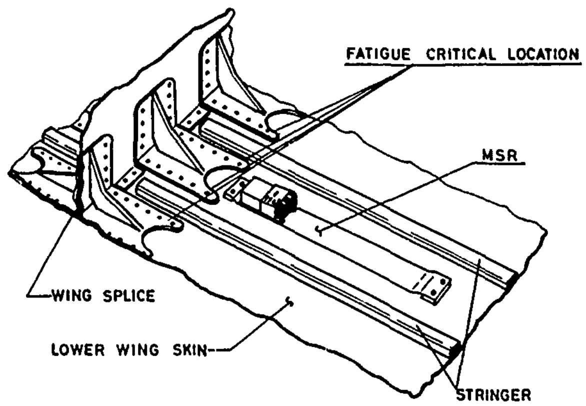
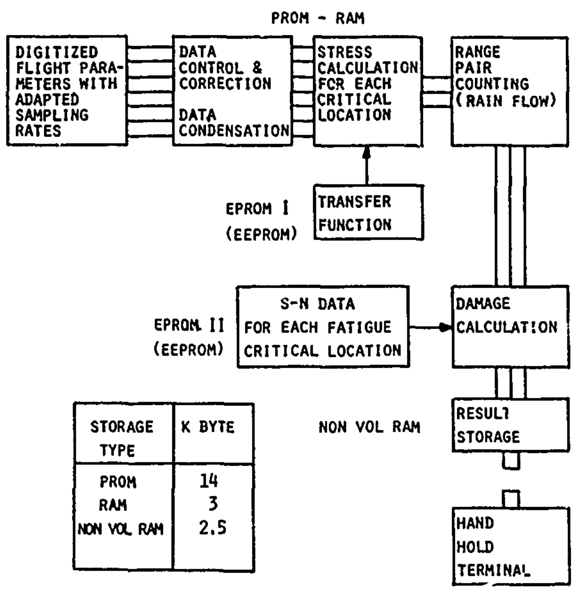


FIG. 10 INSTALLATION OF MSR ON A WING STRUCTURE



STORAGE TYPE	K BYTE
PROM	14
RAM	3
NON VOL RAM	2.5

FIG 11 SUBROUTINES FOR AN ON BOARD DAMAGE CALCULATION SYSTEM

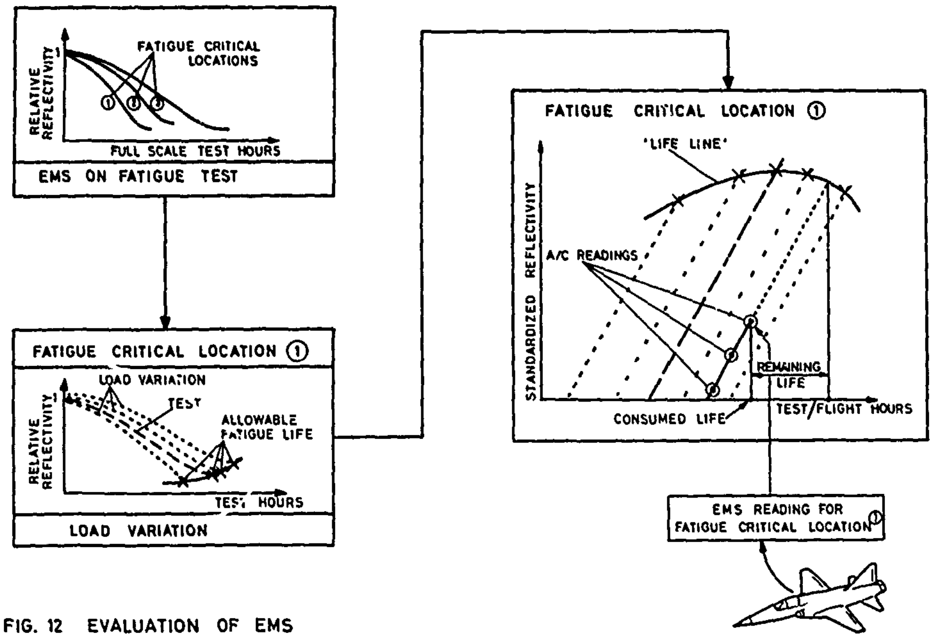


FIG. 12 EVALUATION OF EMS

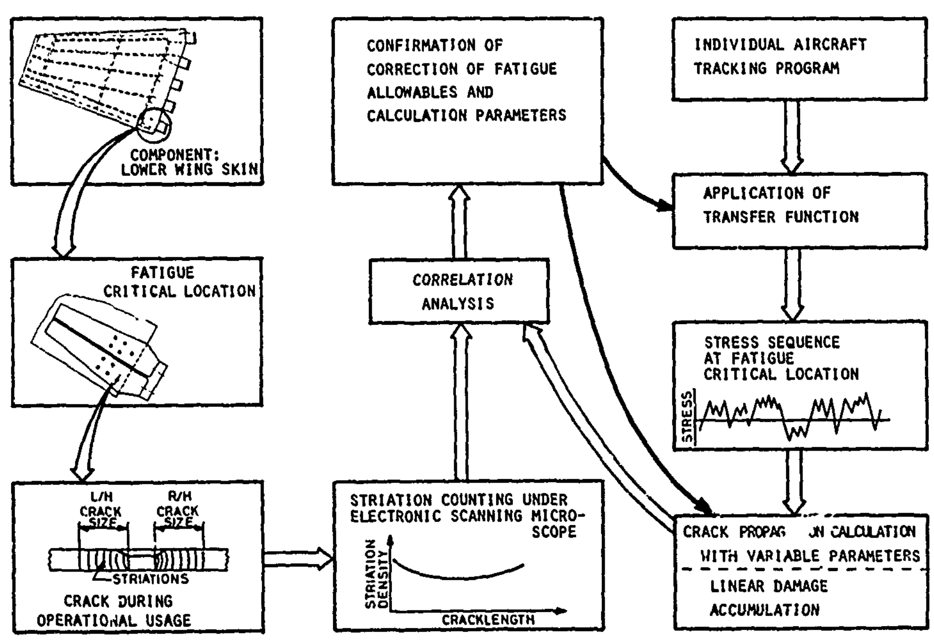


FIG. 13 VERIFICATION OF FATIGUE LOAD MONITORING

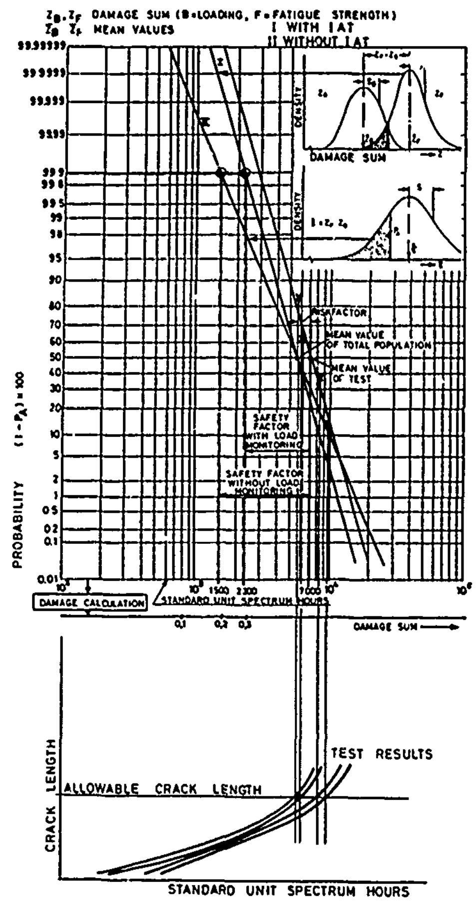


FIG. 14 DETERMINATION OF SAFETY FACTORS

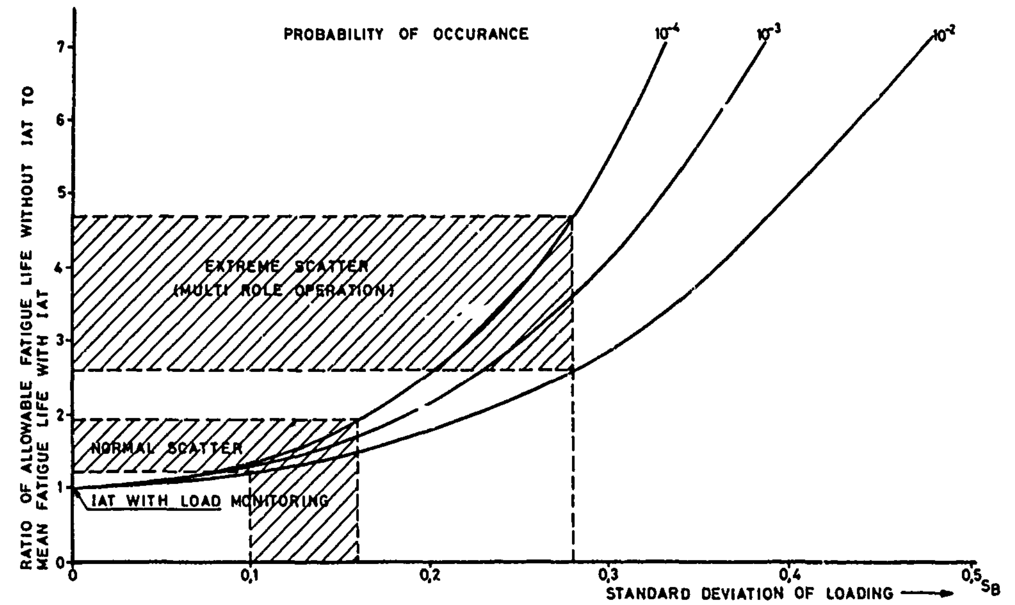


FIG. 15 LIFE SAVINGS DUE TO INDIVIDUAL AIRCRAFT TRACKING (IAT) WITH 4 TEST SPECIMEN

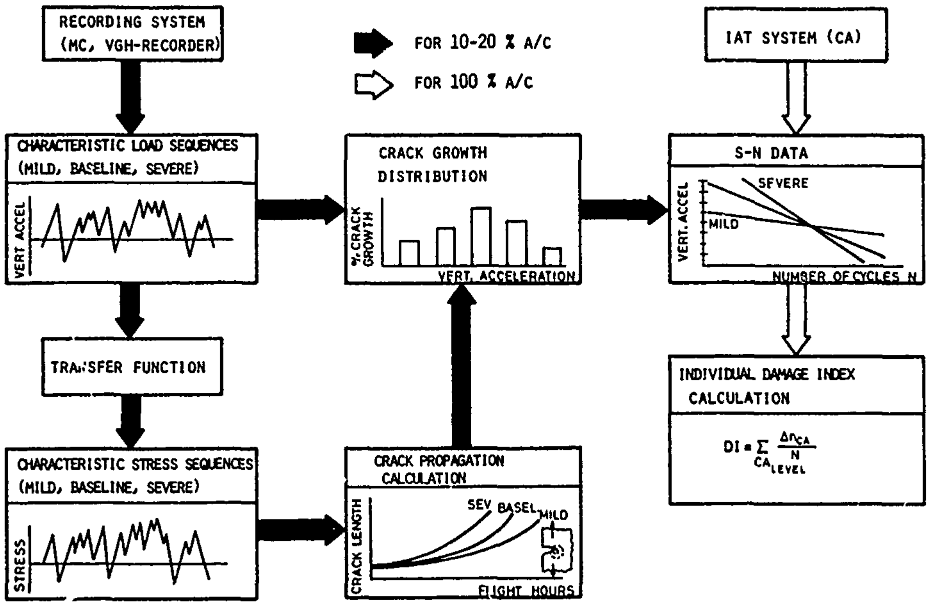


FIG 16 DAMAGE INDEX CALCULATION FOR DAMAGE TOLERANCE STRUCTURES

△3 TYPE	DATA ACQUISITION SYSTEM	DATA VOLUME	EVALUATION VOLUME	COVERED STRUCTURAL AREAS	COSTS HARD-WARE	CALC. PROGRAM	TYPE OF SYSTEM FOR IAT
	FLYING LOG	HIGH	HIGH	MEDIUM	NOTHING	MEDIUM	PRIMARY
FPM	G-METER	LOW	LOW	SMALL	LOW	LOW	PRIMARY
	VGH-RECORDER	MEDIUM	MEDIUM	MEDIUM	MEDIUM	MEDIUM	ADDITIONAL
	MULTICHANNEL RECORDER (MR)	HIGH	HIGH	LARGE	MEDIUM	HIGH	ADDITIONAL
DSM	MECHANICAL STRAIN RECORDER (MSR)	LOW	MEDIUM	SMALL △1	VERY LOW	MEDIUM	PRIMARY & ADDITIONAL
OBDCS	ON BOARD DAMAGE CALCULATION SYSTEM △2	LOW	LOW	LARGE	HIGH	LOW	PRIMARY
LCS	EMS △2	LOW	LOW	LARGE △1	VERY LOW	VERY LOW	PRIMARY

△1 MAINLY DEPENDENT ON ACCESSABILITY　　△2 INSTALLATION PLANNED

△3 FPMS = FLIGHT PARAMETER MONITORING SYSTEMS
DSM = DIRECT STRAIN MONITORING SYSTEM
OBDCS= ON BOARD DAMAGE CALCULATION SYSTEM
LCS = LOAD CUMULATION SYSTEMS

FIG 17 COMPARISION OF APPLIED DATA ACQUISITION SYSTEMS

STRUCTURAL FLIGHT LOAD MEASUREMENT
DEMONSTRATION OF STRUCTURAL INTEGRITY

by
E. Rauscher
Flight Test Engineer, Airbus Program
Messerschmitt-Bölkow-Blohm GmbH
Toulouse, France

ABSTRACT

In this paper structural flight load testing is reported. Procedures presented as an example here were used for flight testing on different types as fighters and military and civilian transport aircrafts.

It is described to obtain flight loads from calibration methods including strain gauge bridge selections. There are several evaluation methods for short and long flight periods to check design loads for static and fatigue criteria. The Maximum Likelihood method is used to investigate aerodynamic coefficients. Counting procedures are used for statistical purposes.

I. INTORDUCTION

Load measurements serve to check for adequate dimensioning of static and dynamic load cases as well as establishing critical loads, which are not adequately covered by relevant specifications. Such measurements are a requirement of military and civilian specifications and regulations.

The MIL-A-8871 specifications require for certification structural flight load testing in a very complete description and a lot of measurements, e.g. wing load distributions of shear, bending, torsion etc.. This specification covers also flight test requests.

Of late, FAR part 25 § 25.301 requires such flight load tests for commercial aircrafts. But additional to the airworthiness requests, manufactures of military and civilian aircrafts are interested in development tests for loads, stresses etc.. This closes a logical circle, which, depending on the progress of the dimensioning and/or certification phase consists of: model testing, determination of input data for calculation of sectional loads, static and dynamic tests and finally, control in form of flight testing incorporating the determination of the stationary and dynamic behaviour of the overall aircraft and its components.

For checking of the assumed load spectra for the fatigue test, long-time inservice measurements are taken. The measured data (component loads, accelerations deflections of rudder etc.) are assessed on the basis of statistical procedures and thus, the actual load spectra derived. These results are taken as a basis for a comparison with and/or the required correction of the assumed load spectra.

An analysis of these load measurements in a positive case allows interference in frequently very costly major tests to achieve economical corrections. In the opposite case, i.e. with a prevailing negative result, the analysis facilitates the elimination of weaknesses prior to commencement of production.

Additionally, information is available which will provide more exact flight computer, simulator, control unit and flight control system inputs, this acquiring relevancy for direct lift control, manoeuvre load control, gust reduction etc.

These load measurements are based on strain gauges installed in the aircraft. The gauges permit precise load measurements when they are sensibly arranged and calibrated.

The incorporation of data describing the flight condition (movement parameters) enables an analysis of load portions.

II. CALIBRATION PROCEDURE AND STRAIN GAUGE BRIDGE SELECTION

Calibrated strain gauges are commonly used to obtain flight loads. They are installed at those places of the structure, which are assumed to show linear relationship towards loading. As a rule shear bridges are bonded to spar webs, bending bridges onto spar flanges or stringers and torsion bridges onto the skin. The load calibration will be performed by applying discrete loads in a grid pattern over the surface. Strain outputs $\mu = \frac{\delta}{\delta_{cal}}$ as a nondimensional gauge response due to load will be recorded. So a load equation can be developed in the following form:

$$L = [\mu_1 \ \mu_2 \ \mu_3 \ \ \mu_j] \begin{Bmatrix} \beta_{11} \\ \beta_{12} \\ \beta_{13} \\ \vdots \\ \beta_{1j} \end{Bmatrix}$$

L is the load
ß is the influence coefficient

In general form the calibration procedure is described in NACA Report 1178. [1]

A rectangular matrix system is generated, whose load vectors may alternately be shear, bending or torsion. The following example contains n loads and j strain bridge outputs.

$$\begin{Bmatrix} L_1 \\ L_2 \\ L_3 \\ \vdots \\ L_n \end{Bmatrix} = \begin{vmatrix} \mu_{11} & \mu_{12} & \mu_{13} & \cdot & \mu_{1j} \\ \mu_{21} & \mu_{22} & \mu_{23} & \cdot & \mu_{2j} \\ \mu_{31} & \mu_{32} & \mu_{33} & \cdots & \mu_{3j} \\ \vdots & \vdots & & & \\ \mu_{n1} & \mu_{n2} & \mu_{n3} & \cdots & \mu_{nj} \end{vmatrix} \begin{Bmatrix} \beta_{11} \\ \beta_{12} \\ \beta_{13} \\ \vdots \\ \beta_{1j} \end{Bmatrix}$$

The solution of this overdetermined equation system $n > j$ is conditioned by the non-linearity of measured values, i.e. this equation system is solved for $\{\beta_{1j}\}$, according to the method of least squares. Thus an influence coefficient is derived for each bridge.

Retroactively, a control vector for each calibrated load of the measured load can be calculated from solution β. For this, the difference between applied and measured calibration is derived as follows:

$$\{\varepsilon_v\} = \{L\} - \{L'\}$$

Thus, the following probable error of the load vector results:

$$P.E\,(L) = 0.6745 \sqrt{\frac{\Sigma \varepsilon_v^2}{n-(j-1)}}$$

Error estimation of the influence coefficients for each bridge are achieved by using terms (variances) of the main diagonal of the following matrix:

$$\begin{bmatrix} m_{11} & m_{12} & m_{13} & \cdot & m_{1j} \\ m_{21} & m_{22} & m_{23} & \cdots & m_{2j} \\ m_{31} & m_{32} & m_{33} & \cdot\cdot & m_{3j} \\ \vdots & \vdots & \vdots & & \\ m_{j1} & m_{j2} & m_{j3} & \cdots & m_{jj} \end{bmatrix} = \left[\| \mu_{nj} \|^T \| \mu_{nj} \| \right]^{-1}$$

From this, the deviations of the β - values are obtained:

$$\begin{Bmatrix} P.E.(\beta_{11}) \\ P.E.(\beta_{12}) \\ P.E.(\beta_{13}) \\ \vdots \\ P.E.(\beta_{1j}) \end{Bmatrix} = P.E.(L) \cdot \begin{Bmatrix} \sqrt{m_{11}} \\ \sqrt{m_{22}} \\ \sqrt{m_{33}} \\ \vdots \\ \sqrt{m_{jj}} \end{Bmatrix}$$

P.E. (β) must be understood as a scatter value of the coefficients. Thus redundant and irrelevant bridges can easily be detected and sorted out. However, during the frequently time - critical calibration phase it would be too troublesome, to manually prepare a new combination of matrices subsequent to the solution of the first equation system and the first P.E. (β). This will never be optimal.

The evaluation of the calibration will be done by computer program automatically:

All bridges are always incorporated into the matrix to determine β- values and error - outputs. Subsequently the program generates the quotient $\frac{\text{BETA}}{\text{P.E. (BETA)}}$.

Thus sorting out the worst response column, in which this quotient holds the smallest value. In doing so, the first "bad" bridge is cancelled and the procedure automatically commences with a new, smaller measuring value matrix. This procedure is applied for all load vectors until only one bridge is left over.

The error of the load is by no means at a minimum when incorporating all bridges. On the contrary it rises upon sorting out of a few measuring value columns and rises again when a small number of bridges prevails. (See figure 1)

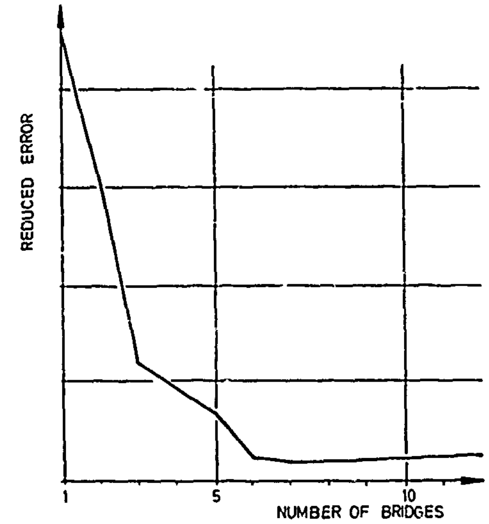


FIG. 1 ERROR DUE TO BRIDGE ELIMINATION

When selecting the bridges the decision should rather be made in favour of a small number of bridges with a reasonably acceptable error, since most measuring points are no longer accessible after failure. Even in case of a bridge failure new combinations which can be calculated beforehand can be prepared.

So the optimal bridge selection can be carried out in regard to load error and number of strain gauge bridges. The realization can be done either in generating electrical bridge summation circuits or by recording single bridge responses and using computer to add those signals to obtain pure load measurements.

Electrical combinations have been successfully used in flight load testing of military aircrafts TRANSALL C160, VAK 191B, TORNADO undercarriages and in the aircraft VFW 614.

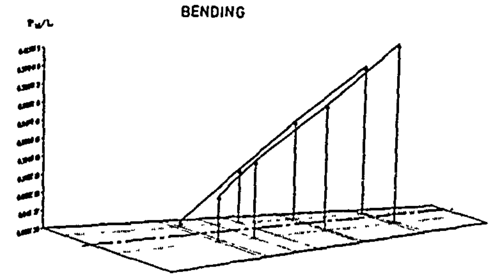


FIG. 2 INFLUENCE COEFFICIENT PLOTS
A300-600 FIN CALIBRATION

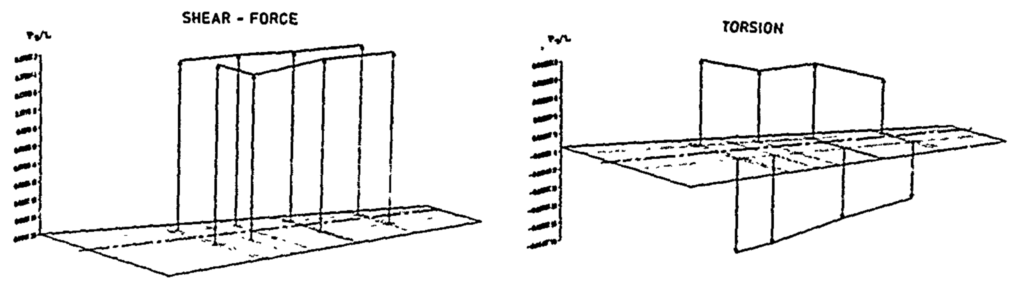


FIG.2 INFLUENCE COEFFICIENT PLOTS
A300-600 FIN CALIBRATION

III. SURVEY OF EVALUATION TECHNIQUES AND EXAMPLES

In order to check assumptions and tests relating to the structure, evaluations must be carried out for the entire frequency range ($0 \leq f \leq 5000$ Hz). In the case in question, a distinction is made between dynamic and static problems.

To permit an assessment of service life issues on the basis of flight tests, it is imperative that long-term measurements be performed with definite load parameters. Such long-term measurements must cover a period of at least one year to fulfill certain statistical safeties and to make allowance for seasonal meteorological influences. Moreover, the data are to be acquired in scheduled service in order to obtain information on characteristics typical for service conditions.

Statistical counting methods are employed to evaluate such a large quantity of data. Detailed descriptions of this are given in the references. [10], [11]

Standard methods worth mentioning:

- range pair
- level crossing
- peak counting
- rain flow

These are one-dimensional methods describing the measured parameter as a function of the frequency.

Two-dimensional methods are also applied, showing the dependence of two parameters as a function of the frequency. (Example: connection between vertical tail load and the roll moment at the horizontal tail or the landing gear loads in forward and lateral direction).

Long-term measurements involve a minor test scope with reference to the life of an aircraft. Extreme value distributions are used for extrapolation of the evaluated spectra with regard to the life. Unfortunately, such evaluations of scheduled flights are only available very late, so that it is necessary to perform such investigations beforehand during flight tests. Many ground taxi runs and ferry flights to test locations can be used for this purpose.

In point of fact, the result of long-term measurements serves to check out the loads assumed for fatigue. Corrective measures can be introduced into the demonstration calculations and tests. Application to aircraft variants is very helpful.

From the point of view of timing, the check of static demonstrations is more favourable. Structural flight tests should commence early on in the flight testing phase. Corresponding regulations give definitions of the flight conditions with which the load level is checked. In contrast to the civil regulations FAR 25 § 301, the MIL specification of the structural flight tests, detailing the components and locations at which loads should be measured. This specification gives basic values of the flight spectrum (e.g. Mach number, altitude combinations) as well as defined abrupt manoeuvres with the aim of reaching a predetermined load level. Prior to completion of the static laboratory tests, 80 % of the limit load and after the test 100 % of the limit load must be demonstrated in flight. Extrapolation to the limit load should here take place at an early stage.

The aim is to disturb the dynamic system, i.e. the aircraft in such a way that a clear answer can be evaluated. An abrupt short manoeuvre is very unsuitable for this. Although a high load level is reached for a short time, the dynamic informatory content is small. Control surface inputs which excite the aircraft to such an extent that both the rigid-body motion and phygoids are clear, are more favourable.

Extensive examinations are detailed in references [8]. We have decided in favour of the multi-stage signal that can be controlled by the pilots.

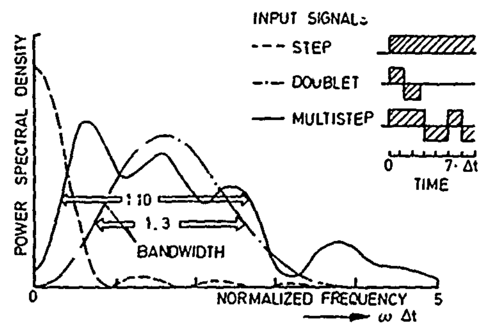


FIG.3 FREQUENCY DOMAIN COMPARISON OF VARIOUS INPUT SIGNALS see Ref 8

After some practice, the pilots were able to apply the control surface inputs so that the power spectrum of the disturbance is Satisfactorily complete.

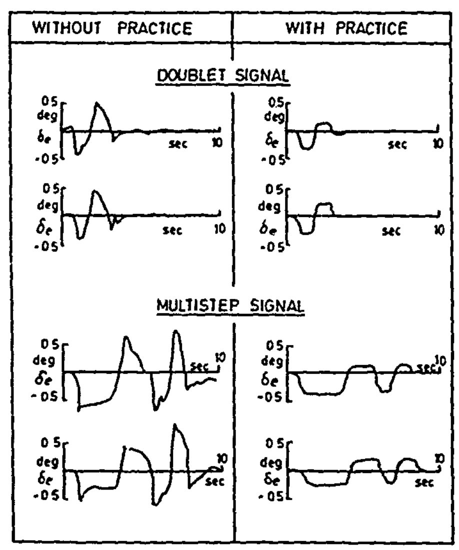


FIG.4 PILOT FLOWN INPUT SIGNALS see Ref.8

Aerodynamic coefficients obtained by way of wind tunnel measurements and empirical procedures serve as a basis for dimensioning in the design phase. An important step in the certification phase during flight testing is to check these values.

In the past, so-called digital matching procedures were used by us to recalculate the flight load parameters. The measurement and calculation were adapted by manually manipulating the aerodynamic coefficients entered into the mathematical model. This procedure is uneconomical because it requires much time and requires considerable experience on the part of the user.

Recently, the Maximum Likelihood procedure has been used. Details will, however, not be given on this procedure here as references provide an extensive description of it [7].

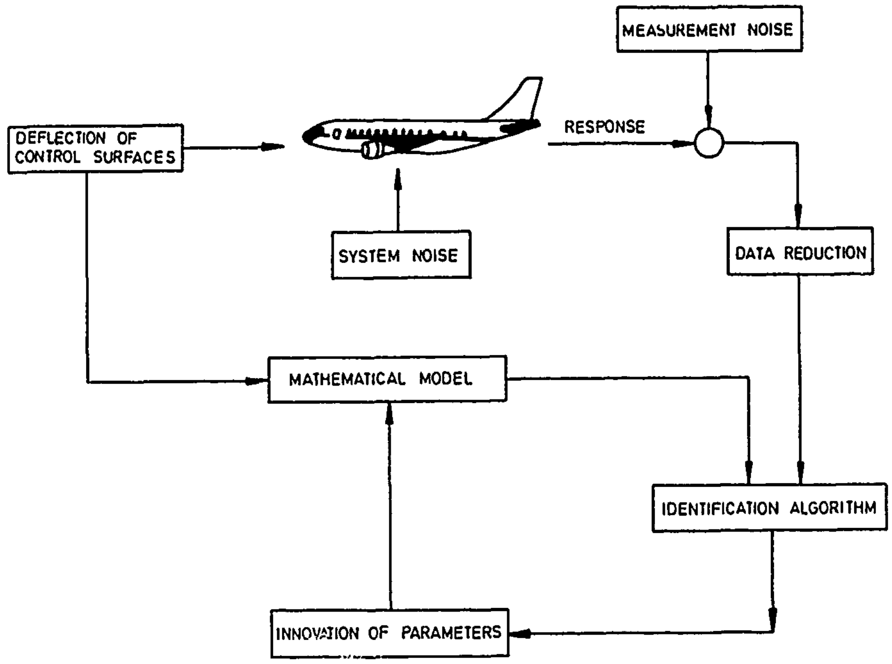


FIG.5 FLOW CHART OF PARAMETER IDENTIFICATION

Based on the good cooperation with the DFVLR (German aerospace research and test institute) in Braunschweig and Oberpfaffenhofen, we have adapted computer programs which are used with and without Kalman Filters. With these it is possible to eliminate both measurement and system noise. These programs have been adapted to our purposes. In other words, we have added component load equations to the system output. Parameter identification was successfully achieved on the strength of the component load measurements:

- horizontal tail load
- wing load
- vertical tail load

The following gives an example of the system of initial magnitudes for longitudinal movement; the measurement equation set contains the horizontal tail load:

state equation

$$\frac{d}{dt}\begin{bmatrix} u \\ \alpha \\ q \\ \vartheta \end{bmatrix} = \begin{bmatrix} X_u & X_\alpha & 0 & -g \\ Z_u & Z_\alpha & 1 & 0 \\ 0 & M_\alpha & M_q & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix} \cdot \begin{bmatrix} u \\ \alpha \\ q \\ \vartheta \end{bmatrix} + \begin{bmatrix} 0 & X_{\alpha^2} & 0 & 0 & 0 & X_\Delta \\ Z_{\delta q} & 0 & Z_{\dot\alpha} & Z_{\dot q} & Z_{nz} & 0 \\ M_{\delta q} & 0 & M_{\dot\alpha} & M_{\dot q} & M_{nz} & M_{\Delta z} \\ 0 & 0 & 0 & 0 & 0 & 0 \end{bmatrix} \cdot \begin{bmatrix} \delta q \\ \alpha^2 \\ \dot\alpha \\ \dot q \\ n_z \\ \Delta s \end{bmatrix}$$

measurement equation

$$\begin{bmatrix} a_x \\ a_z \\ \alpha \\ q \\ \vartheta \\ P_{HT} \end{bmatrix} = \begin{bmatrix} X_u & X_\alpha & 0 & 0 \\ 0 & V_\infty Z_\alpha & V_\infty Z_q & 0 \\ 0 & 1 & 0 & 0 \\ 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 \\ 0 & Z_{\alpha h} & Z_{qh} & 0 \end{bmatrix} \cdot \begin{bmatrix} u \\ \alpha \\ q \\ \vartheta \end{bmatrix} + \begin{bmatrix} 0 & X_{\alpha^2} & 0 & 0 & 0 & X_{\Delta s} \\ V_\infty Z_{\delta q} & 0 & V_\infty Z_{\dot\alpha} & V_\infty Z_{\dot q} & V_\infty Z_{nz} & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 \\ Z_{\delta qh} & 0 & Z_{\dot\alpha h} & Z_{\dot q h} & Z_{nzh} & 0 \end{bmatrix} \cdot \begin{bmatrix} \delta q \\ \alpha^2 \\ \dot\alpha \\ \dot q \\ nz \\ \Delta s \end{bmatrix} + \begin{bmatrix} W_{ax} \\ W_{az} \\ W_\alpha \\ W_q \\ W_\theta \\ W_p \end{bmatrix}$$

Figure 6, shows the adaptation of flight measurement and calculation. The wind tunnel parameters always served as initial values for the calculation. The result, particularly that of the components, was fairly good.

Generally speaking, it can be said that coefficients applicable to the overall aircraft can also be determined better by including load measurements.

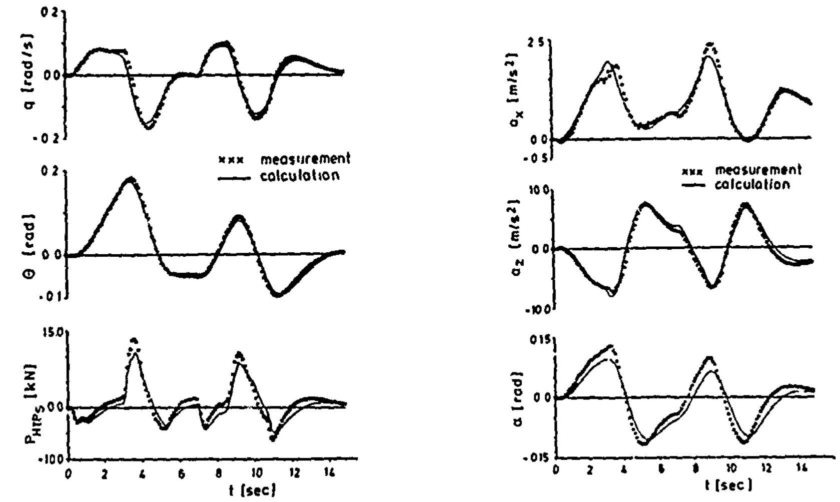


FIG.6 COMPARISON OF MEASURED AND CALCULATED DATA LONGITUDINAL MOVEMENT

These short time measurements are used to recalculate static dimensioning load cases. That means evaluations and verifications over the range of Ma, altitude and dynamic pressure are extrapolated to the static design load cases.

Concerning in flight load measurements for fatigue at the TRANSALL (C 160 A-04) extentive measurements were taken in military training operations and in service life. (C160 D41).

Exemples given in this report were selected [Ref. 6] from service life measurements of rudder-, aileron- and elevator deflections. Additional to this also speed and altitude were recorded.

Different statistical counting methods were applied for evaluation purposes to demonstrate the correlations of one parameter to each other. Also diagrams show the coherence of moving surface deflections to the dynamic pressure which gives the information of loadings. These joint distributions are shown on FIG. 7 bis 11.

Exploitations were reduced to 1 hour, unit deflections $\frac{\text{actual deflection}}{\text{max deflection}}$ and referenced to dynamic pressure at V_D

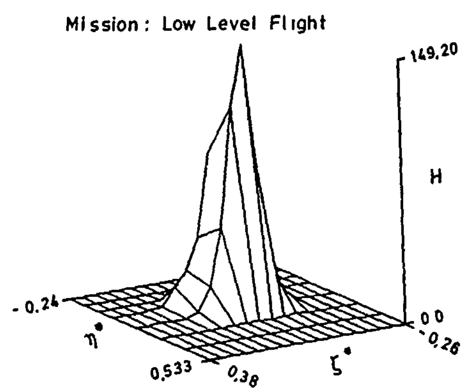


FIG.7 STATISTICAL CORRELATION PLOTS

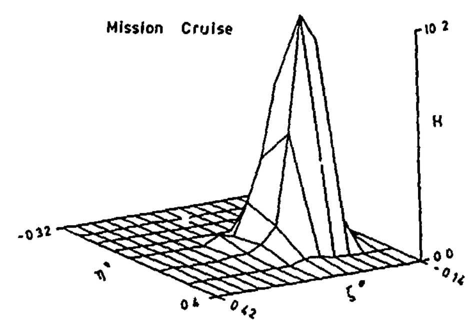


FIG.8 STATISTICAL CORRELATION PLOTS

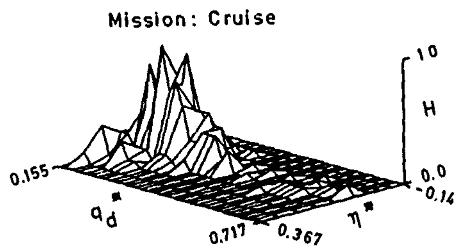


FIG.9 STATISTICAL CORRELATION PLOTS

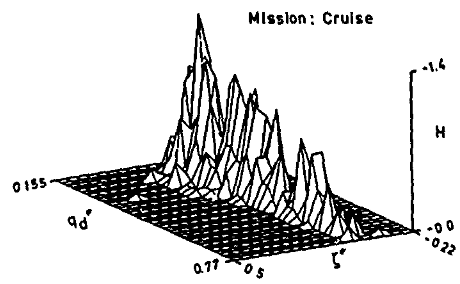


FIG.10 STATISTICAL CORRELATION PLOTS

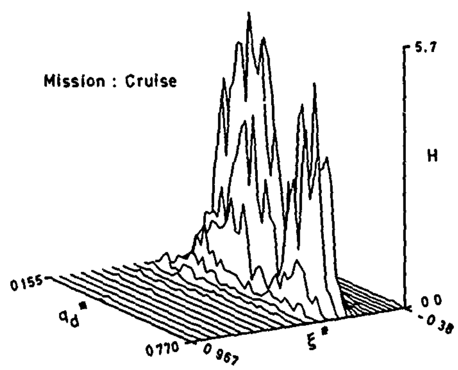


FIG.11 STATISTICAL CORRELATION PLOTS

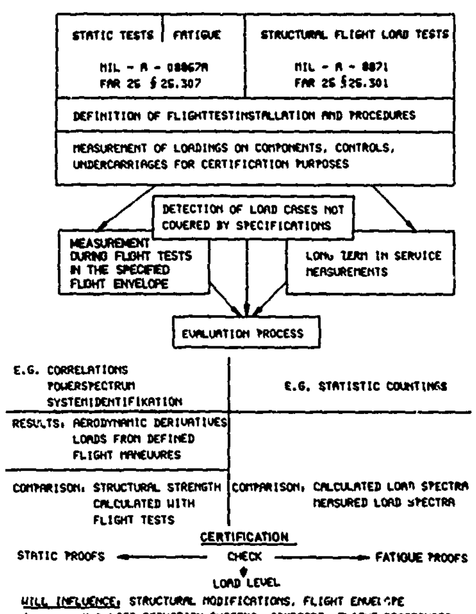


FIG.12 FLOW CHART OF STRUCTURAL FLIGHT LOAD TESTING

IV. CONCLUSION

Structural flight load testing is an important part of the certification of an aircraft. This report describes calibration of strain gauges and bridge selection and evaluation of flight parameters, for fatigue and static test problems.

For aerodynamic parameter identification an optimum input is necessary. Maximum Likelihood withKalman Filter is a wellknown, efficient method for parameter identification. In addition to this fatigue evaluations of moving surface deflections in Service flights were shown. Extrapolations from short and long time measurement closes the loop from flight testing to static and fatigue tests.

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SESSION I - OVERVIEWS AND GENERAL PRINCIPLES

SUMMARY RECORD

by
W G Heath
British Aerospace PLC,
Weybridge Division,
Woodford, Cheshire, UK.

Four main topics emerged in the discussion. These were identified as:

Confidence in the system
Accuracy and reliability
Economics of load measurement
Value of on-board processing.

1. Confidence in the System

A clear distinction was drawn between the measurement of loads on aircraft and those on engine components. Engine loads could be more easily related to a datum test, so that life to first crack could be estimated. Airframe fatigue behaviour could only be based on a major test; the representation of the loads on the specimen was of fundamental importance. The more closely the loads resembled service experience, the greater the confidence in the results. It was pointed out, however, that ultimately all fatigue analysis depended on Miner's Law, and no progress had been made towards a more cumulative damage assessment. Scatter was still of the order of 5:1, although large samples from a fleet over long periods of time showed lower levels of scatter.

Confidence also depended on the airborne testing which preceded service flying and on the number of aircraft in a fleet fitted with operational loads measurement (OLM) instrumentation. Correlation with the remaining aircraft was needed on a mission basis.

2. Accuracy and Reliability

Questions were raised regarding the reliability of strain-gauges, which could be in service for long periods. Users' experience was generally very good, even after long periods in storage. More trouble was experienced with recording systems.

There were also queries regarding the accuracy of the Dornier reflectivity gauge, and the difficulties inherent in separating flight loads from the ground-to-air cycle. This gauge's shortcomings appeared to be offset by its cheapness, and by the absence of other simple damage-measurement equipment with reasonable reliability.

Spectrum truncation was noted as a further source of inaccuracy. The errors inherent in calibrating a wing in isolation from its fuselage were also highlighted.

3. Economics of Load Measurement

Provided sufficient aircraft in a fleet could be monitored, OLM could show significant cost savings. It was, of course, necessary to conduct a major fatigue test (costing say £10M), and the cost of simple instrumentation (counting accelerometers) across the fleet might add a further £2M. More comprehensive instrumentation on selected aircraft would increase the cost, but major savings in rework programmes costing between £30M and £100M could be anticipated due to greater accuracy in life calculations.

4. Value of On-Board Processing

As in (1) above, comparisons were drawn between engine and airframe practice. Whilst on-board processing was accepted for engine data, there were two reasons why it was not considered suitable for airframes:

(a) The fatigue test may not be complete, thus preventing the establishment of a reliable S-N curve. Whilst on-board processing could provide an interim result, the range mean pairs analysis should be retained for later evaluation.

(b) Since it was not a 'drop-dead' situation, there was no need to present the pilot or groundcrew with an immediate answer. On-board processing tended to over-simplify the analysis, and there was always time to complete a more thorough analysis on the ground.

ADVANCED FATIGUE MONITORING ON SERVICE AIRCRAFT

by

A Walkden

Structures Department

British Aerospace PLC Aircraft Group

Manchester Division

Chester Road, Woodford
Stockport Cheshire SK7 1QR
United Kingdom

SUMMARY

Advanced fatigue monitoring is currently possible via on-board digital data acquisition systems and microprocessors using measurements of strain or aircraft motion or both. Before embarking on a fleetwide fit of these 'advanced fatigue meters', a comprehensive understanding of aircraft loading and fatigue performance is required.

This paper describes an exercise on an RAF aircraft where typical service data have been collected to a) define the loading actions which should be included in a full scale wing fatigue test; b) enable theoretical fatigue lives for wing, tailplane, fin and undercarriage to be calculated; c) define which parameters might be measured in a comprehensive fit of advanced fatigue meters so that the life of the main structural components of each aircraft could be accurately measured.

1. INTRODUCTION

The fatigue meter has provided, and continues to provide, valuable data from service aircraft. In conjunction with results from full scale or detail fatigue tests, whose load spectra are usually derived from fatigue meter returns, it has given us all an enlightening insight into the problems of realistic fatigue life monitoring. The many unexpected in-service fatigue failures over the years have, however, demonstrated the limitations of both the fatigue meter and fatigue testing based on theoretical load spectra. The current cost of the vehicle, its relative scarcity and the political and economic necessity of a long service life require that for the most cost-effective management of the fleet the fatigue performance of the structure must be more accurately assessed and the accumulation of fatigue damage in each airframe more accurately monitored. This paper describes how we are approaching the problem on the Nimrod fleet.

The fatigue evaluation of any structure requires two pieces of data: the fatigue resistance of the structure and the load spectrum encountered. The Nimrod structure is based on the Comet 4 airliner whose fatigue resistance was established some years ago by a major test with loading presumed typical of that encountered by such aircraft. The operational environment of the military Nimrod is of course vastly different to that encountered by the airliner. Wing life as calculated via a fatigue meter formula based on the early test becomes more marginal as the aircraft develops with increasing AUW and new roles. This state of affairs, coupled with results from an earlier OLM programme on a tanker aircraft where the fatigue meter based formula was subsequently discovered to grossly underestimate wing fatigue damage, led to the decision to carry out a new test on a Nimrod wing. In order to derive representative data for this new test the Nimrod Operational Flight Load Measurement Programme (NOFLoMP) was developed. The aims of the NOFLoMP were: i) to co-ordinate the service vehicle and full scale wing fatigue test specimen by measuring typical loads on the former and then applying these loads to the latter; ii) to provide realistic stress spectra from which the fatigue life of the fleet could be more accurately calculated and give advanced warning of potential fatigue problems in all major structural components based on current assessments of the fatigue resistance of those structures; iii) to help indicate which parameters could be measured in a fleetwide fit of an advanced fatigue meter which would monitor all major structural components.

To achieve these aims a standard Nimrod MR Mk 2 aircraft has been extensively instrumented (Ref 1) and during two years service operation a representative series of sortie profiles have been flown and recorded data are being analysed. The following paragraphs detail the instrumentation and describe the methods of analysis.

2. NOFLOMP - MECHANICS OF THE SYSTEM

2.1 Sampling Rates

A total of 96 strain gauge bridges, 17 accelerometers and 34 general and monitoring parameters were defined as the total instrumentation requirement. Past exercises had demonstrated that measurement of strains in major components of large aircraft for fatigue analysis was satisfactory using 20Hz low pass filters with FM recording. Using this as a basis coupled with a decision that 7 samples per cycle would give good definition led to a maximum sampling rate of 140 per second. This was used for all accelerometers (except those at the aircraft cg) and fin, tailplane and undercarriage strain gauges. The wing strain gauges and other parameters were sampled 70 times per second in conjunction with 10Hz low pass filters. The sampling rate for general parameters (eg speed, altitude) is obviously over generous but, since the chosen system capacity was adequate, maintenance of only two sampling rates led to easier data handling.

2.2 Recording System

Data are recorded on to 25mm 14 track magnetic tape via a Base 10 M series data acquisition unit with High Bit Rate system which writes to the tape in serial/parallel format The tape recorder is an Ampex AR 1700 model which runs at 47.6mm/sec. With the chosen sampling rates, 47 x 10^6 pieces of information are recorded per flying hour and one tape will record about 12 hours flying.

Initially the strain gauges were supplied by a single 12V power pack but it was soon clear that this was unsatisfactory. To prevent corruption or loss of all data as a result of one malfunction 8 power packs were fitted, each supplying some 12 strain gauge bridges.

2.3 Strain Gauge Positions and Load Measurement

2.3.1 Wing

Strain gauges have been attached at four sections on both port and starboard wings for the determination of shear, bending and torque as shown in Fig 1. Preliminary tests on a time-expired Comet at RAE Farnborough enabled sensible positions for these gauges to be determined, and only minor development on the Nimrod was required. From an extensive series of calibration tests, a regression analysis has produced coefficients which when multiplied by strain gauge outputs will determine shear, bending moment and torque. The accuracy of predicting the applied test loads via these coefficients is at worst 5% for shear, 3% for bending moment and 4% for torque at typical 1g flight conditions.

Accelerometers are positioned on each wing at the tip, underwing pod fuel tank, outboard engine rib and aircraft cg to help understand loading actions and enable the inertia loads due to large concentrated masses to be determined.

Aileron and flap loads are deduced by measurement of the control surface actuating rod loads (ie hinge moments) and the application of theoretical load distributions.

Main undercarriage side and vertical loads are measured via strain gauged elements within the undercarriage, and drag load via a reaction member within the wing. The effect on the local wing structure of the undercarriage loads is monitored by measurement of shear in the undercarriage rib and end load strains in the rear spar.

Strain gauges are attached at known fatigue critical areas and also in suitable locations to measure general stresses along the wing. These 'fatigue monitoring' gauges are used to indicate fatigue-damaging events which are required to be reproduced on the fatigue test.

2.3.2 Tailplane

At three sections on the port tailplane and one on the starboard, strain gauges are attached for prediction of bending moment and torque only. Coefficients produced from a regression analysis of a number of tests predict the applied test loads with rather less accuracy than was achieved on the wing. Elevator control rod loads are monitored to give a measure of hinge moments. Fig 2 shows the tailplane strain gauging.

2.3.3 Fin

Gauges were attached to the front and rear spar booms at the root to measure general stresses only, the interaction between main and dorsal fin sections precluding a simple load measuring exercise.

2.3.4 Fuselage/Wing Interaction Loads

Links which attach the wing lower spar booms to the fuselage at the bodyside rib are instrumented and load calibrated. Since lateral bending of the spar booms results from the interaction of wing and fuselage loads via these links, a comparison between such loads on the test specimen and flying aircraft is another aid in ensuring a representative test.

2.4 Data Validity Checks

2.4.1 Verification of Electrical Calibration

As part of the instrumentation switch on and off procedure on the aircraft, a known resistance is applied in parallel with one arm of each strain gauge bridge. On receipt of the computer tape of flight data, on automatic computer check and print out showing strain gauge output with and without the shunt applied together with a list of gauges whose change of output is outside a prescribed tolerance is presented before a decision to proceed with further analysis is taken.

2.4.2 Aircraft on Ground

From the flight data a point is chosen with the aircraft stationary on the ground before and after flight, at known weight and fuel distribution, and a check on the outputs from all channels is taken. For each wing, undercarriage and fuselage strain gauge, an equation has been prepared showing output expected with aircraft weight and fuel distribution. Other channels also have a known state, eg accelerometers at 1g; fin and tailplane outputs at near zero. A computer check, lists expected, measured and error values.

2.4.3 Aircraft in Flight

Test flights have been carried out with the instrumented aircraft when a series of 1g level flight trim points and controlled symmetric and asymmetric manoeuvres were performed over a range of speeds, altitudes, all up weight and fuel distribution. These test flights are an important aid to help verify the flight loads predicted via the loads regression coefficients, and also give datum values for strain gauge outputs in case of doubt with other data verification methods.

2.4.4 Reference Measurements

Bridge and amplifier voltages, together with outputs from reference strain gauges mounted on unrestrained plates, are plotted for each full flight and manually inspected for errors.

3. DATA HANDLING

3.1 Computer System (Fig 3)

The computer system starts with the Pulse Code Modulation (PCM) tape from the aircraft. This tape contains data from 105 sources sampled 70 times per second and 42 sources sampled 140 times per second. The PCM tape will hold about 12 hour's worth of data. The first step in the analysis chain is a validity check on the PCM tape to ensure that the signal strength on each track is sufficient and that the time base is correct so that the replay head will lock on. The tape is then processed to create a set of Computer Compatible Tapes (CCTs). Removing all data from the PCM results in each CCT containing about 20 minutes of flight data so that a 6 hour flight produces 18 or 19 CCTs. This processing is carried out using PDP 11/44 and 11/45 computers.

This first set of CCTs are then fed to a dedicated VAX 11/780 computer and undergo the verification process described in para 2.4.1. The verification indicates if any data channels are totally unserviceable and whether further processing of the CCTs is justified.

Assuming that continuation i, signified, the next stage is the two-rate data compression which produces a second and smaller set of CCTs. At the same time data are checked for various errors and corrected where possible.

Single point drop-outs are dealt with automatically during the data compression stage and are overwritten by the value of the preceeding data.

Interference due to high frequency (HF) radio transmissions has been encountered. Fitting of suitable capacitors to all amplifier inputs has eradicated the interference except at frequencies of 16 and 18MHz. Since the time spent HF transmitting is generally small (a total of 10 minutes in a six hour flight being typical) a monitor has been fitted to indicate when transmissions are taking place. During the compression phase of data handling a mask is introduced at these times so that data are ignored by fatigue analysis programs during the relevant periods.

Data from this second set of CCTs are plotted using a Versatec V80 electrostatic line printer and the results inspected for a final visual channel-by-channel verification. Any remaining spikes, datum shifts, etc have to be found by inspection of the plotted data. If a particular channel contains many spurious points it would normally be regarded as unserviceable and not analysed for that flight. Isolated spurious data are dealt with by inputting the relevant time to a masking program which is then incorporated into a tape copying process to create a replacement CCT from the second set. For security a duplicate set of CCTs is made and stored separately.

3.2 Data Compression

The high sampling rates used on this exercise were intended to ensure that no significant stress peak or trough was omitted from the recorded data. The volume of data thus acquired is, however, far too great to process in full, and the objective of the data compression process is to reduce the data to those which are significant from a fatigue viewpoint.

The concept of data compression is based on a combination of two self-evident ideas i) that fatigue damage (and fatigue test loading cycles) occur during times of structural (or aircraft) activity, and ii) that significant parts of the flight are quiescent. Such behaviour was observed on an earlier exercise on tanker aircraft. Hence arises the idea of deleting or 'compressing' the non-active parts of the flight, by drastically reducing the sampling rate during such periods.

In principle the data compression works as follows (see ''~ 4): on a trace of strain or acceleration, structural activity is associated with the . ._r excursions from the local mean or quiescent level. The computer is programmed to monitor a number of channels (referred to as 'trigger channels') and to search for large excursions. When one is found, the fast sampling rate is switched on (or 'triggered') and remains so until the trigger channel returns to near its local mean.

In practical terms the system is more sophisticated than outlined above, mainly as the result of experience. The following detailed description covers the current state of the art which we feel is a viable system.

As previously stated the fast sampling rates are 70 and 140 samples a second for low and high rate channels respectively. The corresponding slow sampling rates, for use when there is no structural activity, are 1 and 2 samples a second. Thus if a flight were totally quiescent, a compression ratio of 70:1 would be achieved.

The triggering works on a bandwidth concept which in practice has two forms known as 'fixed' and 'floating'. Under the 'fixed' technique, upper and lower limits are set and remain constant (ie 'fixed') throughout the flight; when the trace is outside these limits the fast sampling rate is switched on, being switched off when the trace returns to within the limits. This type of triggering is suitable for accelerometers which have a constant datum irrespective of the flight conditions. Ideally, fin strain gauges would also fall into this category.

The 'floating' trigger (see Fig 5) uses a constant bandwith, but the mean or datum of the bandwidth is variable in that it moves (or 'floats') towards the running average of the trace. Again the fast sampling rate is used when the trace lies outside the bandwidth. The floating is achieved by what amounts to a high pass filter, which

removes the long period drift and reduces the trigger to the fixed type. This floating type of trigger is suitable for all gauges in which there is a low frequency change in the datum and includes most (if not all) wing and tailplane strain gauges. In practice the fin was also treated as a floating trigger.

The choice of trigger channels, bandwidths and time constants in the floating trigger filter are at the discretion of the user. We are currently using the following techniques:

Normal and lateral accelerometers at the centre of gravity - Fixed
Brake rods - port and starboard - Fixed
Fin root bending bridge - Floating
Tailplane root bending bridge - Floating
Top wing surface gauge - Floating
Bottom wing surface gauge - Floating (this is the 'critical section' for fatigue)

One final point to note before considering the results so far, is that the sampling rate is not switched at the moment of triggering. It was decided at a very early stage to sample at the fast rate from 1 second before an 'on' trigger and until 2 seconds after an 'off' trigger. The effects of altering these times have not been investigated.

When using data reduction or compression it is possible that some valuable data will be lost. Hence the percentage loss of fatigue damage was used to measure the validity of the compressed data. The results to date are presented in Fig 6; these are based on our initial work in which we aimed for a zero loss of measured damage. It is evident from the number of CCTs remaining after compression that there are far too many to allow easy analysis, and that the quiescent periods of flight are somewhate less than may have been imagined. Increasing the trigger levels so that damage losses were of the order of 5% suggested that a more reasonable number of 4 CCTs per flight may well result.

The problem was then approached in a different way by deciding to dump data from the PCM tape at a fraction of the recorded rate and assess damage losses relative to the original data. One flight was selected and for the data recorded at 70 samples per second a damage calculation was separately carried out using every second, third, fourth and fifth point. The data recorded at 140 samples per second were separately analysed using every second, fourth and sixth point. Damage losses due to these reduced sampling rates are shown in Fig 7. From the values obtained it is apparent that dumping data from PCM to CCT at ¼ of the recorded rate leads to losses which are not greater than 5%, except for wing gauges (which are significantly affected by undercarriage loads, and then probably only spin-up drag loads at touchdown).

The decision to create CCTs at ¼ the recording rate has, of course, a prime spin-off in that there are now fewer tapes in the first set of CCTs before the data compression process. The original trigger levels were reinstated for the latter process, and two flights were examined in detail for damage calculated at both full recorded rate and ¼ recorded rate, both uncompressed and compressed. Fig 8 tabulates the results which again show that damage losses are generally less than 5% relative to the baseline data.

4. DATA PROCESSING

4.1 Hardware

Using 147 channels, and sampling rates of 70/140 per second, puts the size of the data processing operation at approximately ten times that of the earlier tanker exercise.

The VAX 11/780 appeared a reasonable choice of processor, although from an early stage it was realised that some form of data reduction was needed to make the task more practical.

Our Flight Test Department had considerable experience and knowledge of existing specialist facilities using PCM tapes; hence, it seemed logical to employ this as a replay facility converting the Base 10 PCM tape into CCT for subsequent processing on the VAX computer.

The initial configuration was (see Fig 9):

a) Shared resource of 2 - PDP 11/45 and 1 - PDP 11/44 for conversion of PCM to CCT.

b) Dedicated VAX 11/780
   4-TE16 tape drives (phase encoded 63 bits per mm)
   1-RP06 176 MByte Disc
   1-V80 Plotter

Subsequently the loan of a fifth tape drive has enhanced reliability.

4.2 Software

Software for the conversion of the PCM tape to CCTs is written in PDP 11 macro and is, at present, a straightforward reading/writing process.

Software for the VAX may be thought of as validation and analysis of the data, running under an overall operating environment, which also provides a detailed recording facility to ensure that the appropriate data for a given process is accessed (Fig 10).

The software was developed by a team of five people from both the Structure Department and Computer Services, the former developing the analysis programs and the latter providing operating procedures, plotting routines and tape handling facilities.

Development has been smooth, the VAX playing a vital role in meeting timescales which would have been thought impossible five years ago.

4.3 Operating Environment

The operating environment has been created using Digital Command Language (DCL), the VAX command language.

Utilities have been written in DCL, but, where real numbers required handling, DCL has been supplemented by COIN, the Company-written COmmand INterpreter.

Whilst the VAX 11/780 may be of 'mainframe' category in processor speed, its facility to identify, record and catalogue magnetic tapes is limited.

With processes which could involve up to 6000 CCTs (at present we have 2800) it has been necessary to introduce library-type facilities to make tape handling more feasible.

An 'Operators Guide' describes the overall operation of the system and a comprehensive HELP facility, which supplements the documentation makes operation simple and greatly aids training.

4.4 Analysis

The data which an anlysis process receives should be free of single point drop outs and be marked for any HF noise; in general it is assumed to be 'technically valid data'.

The fatigue analysis methodology will be described in detail later. From a computational point of view the algorithm provided by Susan D Ellis of RAE Farnborough (Raf 2) was expanded to cater for multi-channel processing.

The Loading Analysis program evaluates shear force (SF), bending moment (BM) and torque (T) at the four stipulated stations on the wing. This is a straightforward multiplication process and, working on sixteen channels per wing, presents little problem. The results produced are stored on magnetic tape and may be plotted, as required, using a standard program. (A much modified version of this program will be used to evaluate the loads for the fatigue test rig).

Although the loading program calculation is basically a multiplication process, the gauge coefficients are required and in order to arrive at these several items of software are necessary.

Firstly, a geometric model of the wing was created to enable the definitive BM, SF and T to be evaluated.

Secondly, the base of test results was obtained from selected load calibration tests.

Lastly, the above two variables were combined, and then RAPIER (Ref 3) was employed to evaluate the gauge coefficients for the load program.

RAPIER was obtained, in 1900 Series Fortran, from RAE Farnborough and initially proved on an ICL 1906A Computer. The code was later converted to VAX 11/780, and this version is now in use.

4.5 System Performance

Major processes that have been carried out to date are COMPRESS, PLOT, DUPLICATE, FATIGUE AND MASK. An example of the processing time for a typical long sortie is given

in Fig 11. The average sortie can be processed in about 75% of the time given in the example. Fig 11 also shows the VAX 11/780 mill utilisation for each of the major processes and demonstrates that two of them have to have sole use of the system or result in an increase in processing time.

The processing times required per CCT for the major processes are COMPRESS 20 minutes, PLOT 2 hours, DUPLICATE 20 minutes, FATIGUE 25 minutes and MASK 20 minutes. The PLOT process in fact takes 10 minutes per pass of a CCT. Plotting all parameters results in 12 passes and hence 2 hours per CCT.

5. CALCULATION OF FATIGUE DAMAGE (see Fig 12)

The calculation of fatigue damage, as stated earlier, starts from the compressed or second set of CCTs, and therefore assumes that they contain a true time history of the significant measured strains of each channel. This currently means that data that have been masked due to HF noise and 'spikes' are ignored.

The program is capable of analysing up to 40 channels, which makes it more than adequate to deal with the fatigue monitoring gauges and such other gauges as require analysis from time to time.

The initial control parameters define which channels are to be analysed, the times at which the damage 'to date' is to be output (known as 'phasing') and the start and stop times for the analysis. These last two control parameters were introduced as a convenient way of avoiding the shunts (pre take-off and post landing) being included in the analysis.

The program then extracts the turning points of each channel and the times at which they occur. These data are in turn fed to a routine which by means of the Rainflow technique extracts the fatigue cycles in terms of a range and a mean. The software to achieve this is a virtual copy of the RAE routine described in Ref 2. The turning points that do not form full fatigue cycles are stored until the end of the flight (or phase) when they are dealt with using the re-ordered stack method which ensures full ground-to-air fatigue cycles. Track is kept of the time by passing them through the Rainflow routine in parallel with their associated turning points. The only control data required are a range for each channel below which a cycle is ignored. Currently 6.9MPa is being used.

The ability to generate a 2-D counted array of the fatigue cycles exists but is not used at present. The ranges and means still have a digitised form and are converted into stresses by a linear equation (σ = m*DRO + c). The coefficient m and the constant c are derived from measurements and calculations, and are held in the computer on a file which is accessed directly by the damage program. The file of coefficients is updated as required. The calibration data is chosen automatically on the basis of channel number, flight number and date by the program.

Having obtained the fatigue loading in terms of stress, the calculation of damage is straightforward, using Miner's Cummulative Damage Law with a Gerber-Goodman mean stress correction. The control data enables a choice of S-N curve to be made. It also supplies the mean stress correction coefficients and two factors on stress: an R value applied to the alternating stress only, and a K value that is applied to both alternating and mean stresses. Usually Heywood's light alloy joint endurance curve is used, i.e. where applicable data from the Comet full scale test are read across.

The full program takes about 25 minutes per tape to run, and the output, which gives details of all the fatigue damaging cycles, is of manageable proportions. Plots of damage against flight time are produced. Currently the damage is plotted cycle by cycle so that the time of peak or trough (or earliest or latest time) of the paired turning points are plotted. Some typical samples are shown in Figs 13 to 15.

The program has been used to establish the triggering levels (outlined in para 3.2) so that the percentage loss of damage due to compression is acceptable. The losses generally involve the missing of peaks of the higher frequency loads, thus truncating fatigue-damaging cycles.

The program defines and files the times at which fatigue-damaging events occur so that the loads producing these events can be derived for application to the fatigue test specimen. The latter will be strain-gauged in a similar manner to the flying aircraft, and data from sample test flights will be analysed via the same program to give a final cross-check between the service and test vehicles. It is envisaged that the damage program will be used to establish the damage per flight or phase against which the Nimrod MR Mk 2 fatigue meter formula will be checked and modified if necessary.

6. CONCLUSIONS

An extensive instrumentation package such as that fitted for the NOFLoMP is obviously expensive to install and maintain, and even with the use of a dedicated powerful computer, time consuming in the analysis of data. In the final analysis a much reduced number of strain gauges and/or basic motion parameters will be identified as key items by which the accruing fatigue damage of each major component of a Nimrod MR Mk 2 airframe could be monitored. It is likely that this will result in a requirement for a new airborne fatigue monitoring device rather than the revised fatigue meter formula approach used as a result of the earlier tanker exercise.

The analysis techniques currently applied rely on Miner's Damage Law, and this is probably the weakest link in the chain. However the more representative the fatigue test loading can be, the less the error introduced by this step in the analysis. The integration of a load measuring exercise on a service aircraft and the full scale fatigue test would appear to be essential. Applying the measured sequence of service loads to the test specimen and, as backup, monitoring the test stress spectra at the fatigue critical locations instrumented on the flight vehicle should remove doubts as to the authenticity of the test result.

The results of the earlier Comet test are being used to provide provisional information on the fatigue resistance of the structure. An initial estimate of fatigue life consumption can thus be made from the loads measurement exercise and give advanced warning of potential trouble spots and timescales. The new fatigue test using a more representative structure and loading actions will then provide a revised estimate of the fatigue resistance, allowing an adjustment to be made to fatigue damage estimates.

A smaller exercise is to be carried out on a Nimrod AEW Mk 3 aircraft. Fatigue monitoring strain gauges are fitted at positions highlighted as significant by the earlier exercise on the MR Mk 2 aircraft, and damage will be calculated based on the result of the Mk 2 wing fatigue test. A requirement for a fin fatigue test has been identified for the MR aircraft, and again a read-across to the similar type of structure on the AEW aircraft is anticipated.

For the future the MR NOFLoMP aircraft will continue to supply data via its existing system until perhaps 1986. It is then intended to fit a SUMS type recorder utilising a small number of strain gauges to monitor any changed usage of the aircraft. Fleetwide fitment of an advanced fatigue meter (eg SUMS) will be assessed as more data becomes available from both fatigue tests and NOFLoMP.

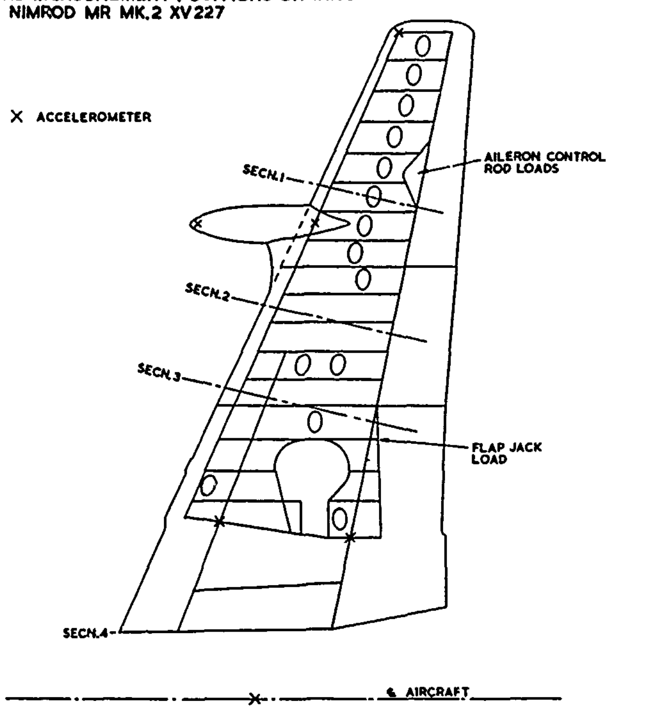
7. REFERENCES

1. AGARD report 704 (September 1982)
   NOFLoMP - 'The Nimrod Operational Flight Load Measurement Programme' by W G Heath

2. RAE Technical Report 81122 (October 1981)
   'A combined range-mean-pairs rainflow count of load time histories for use in the formulation of structurally relevant cost functions and fatigue analysis' by Susan D Ellis

3. NASA Technical Note D-5656
   'Rapier - A Fortran IV program for multiple linear regression analysis providing internally evaluated remodelling' by S M Sidik and B Henry

Fig 1
LOAD MEASUREMENT POSITIONS ON WING
NIMROD MR MK.2 XV227



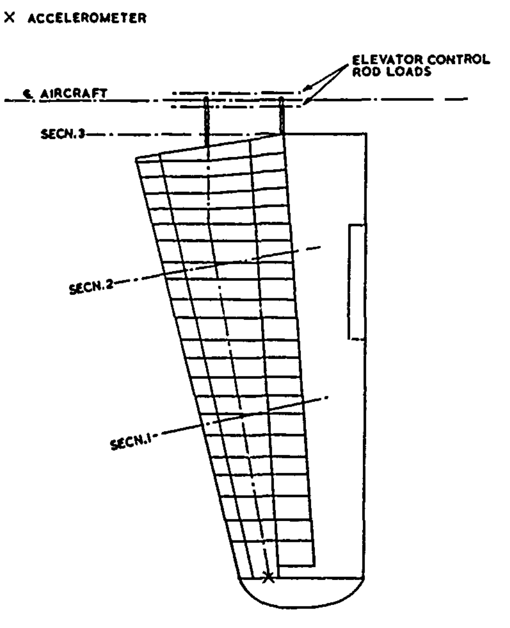
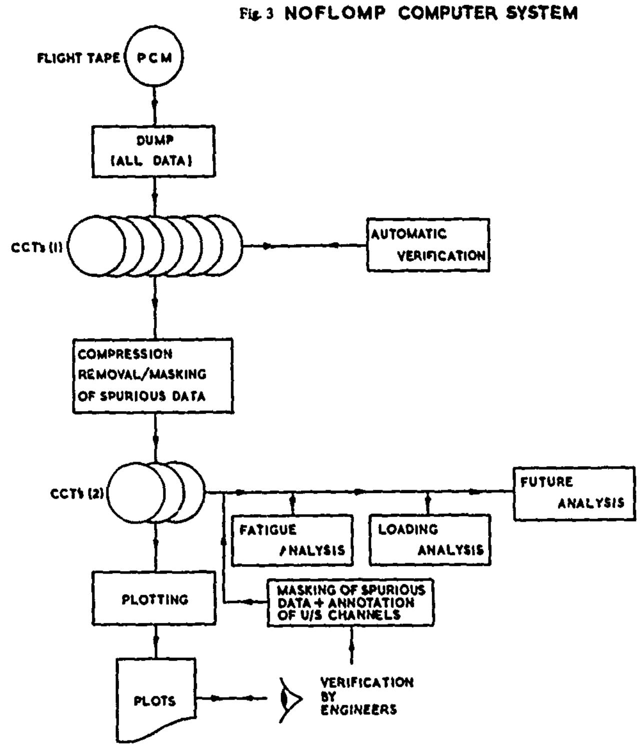


Fig. 2 LOAD MEASUREMENT POSITIONS ON TAILPLANE

Fig. 3 NOFLOMP COMPUTER SYSTEM



4 FIXED TRIGGERING OF HIGH DATA RATE

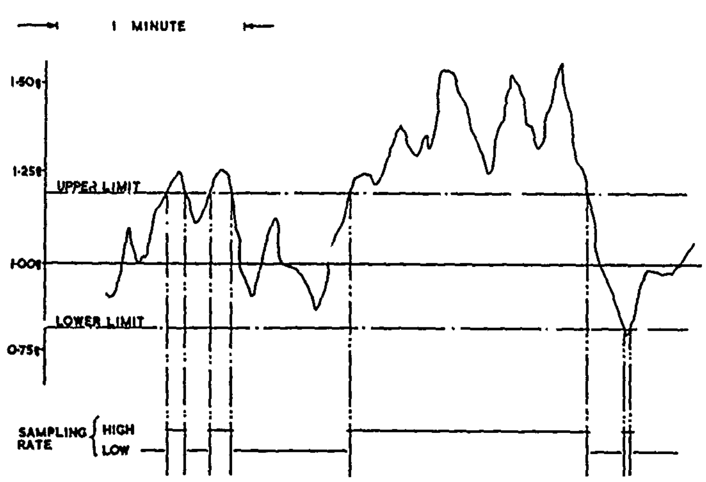
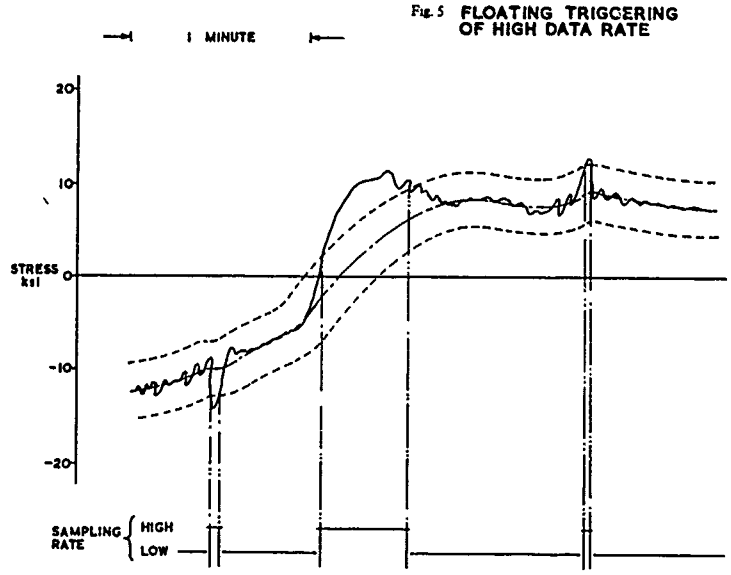


Fig. 5 FLOATING TRIGGERING OF HIGH DATA RATE



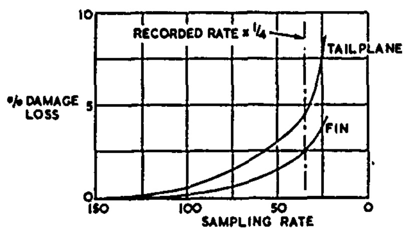
FLIGHT No.	% LOSS OF DAMAGE	DURATION HOURS	No. OF TAPES		ACTUAL COMP. RATIO
			PRE	POST	
102 (Part)	< 0·2	2·83	9	4	2·48
108	< 0·2	4·50	14	10	1·47
115	Not known but triggering is the same as for flights 108 & 102	9·17	26	12	2·31
124		12·08	38	5	8·09
126		9·67	30	9	3·49
130		9·00	27	10	2·87
131		7·50	24	11	2·18
133		9·17	28	10	2·77
139		8·17	26	14	1·88
141		6·58	21	7	3·33
			Σ 234*	88*	Av. 2·66*
108	5		14	4	4·0
ALL EXCEPT 102	5				Av. 7·24Δ

* EXCLUDES FLIGHT 102

Δ ASSUMES RATIO OF COMPRESSION ACHIEVED FOR 108 MAINTAINED FOR ALL

Fig. 6 ACHIEVED DATA COMPRESSION

Fig 7 % LOSS OF DAMAGE WITH VARIOUS SAMPLING RATES
Flight 108 Data



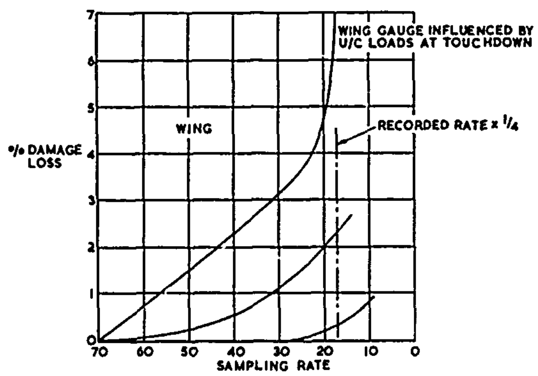


Fig 8 % DAMAGE LOSS RELATIVE TO FULL RATE UNCOMPRESSED DATA

LOCATION	% DAMAGE LOSS FLIGHT 108			
DATA	Full Rate		Quarter Rate	
	Uncompressed	Compressed	Uncompressed	Compressed
Compression Ratio	1·00	1·47	4·00	5·86
TAILPLANE	0	0·20	3·85	4·35
FIN	0	0	2·60	2·60
OUTER WING	0	0	1·85	2·78
INNER WING	0	0	1·32	1·32
INNER WING + U/C	0	0	7·20	7·20

LOCATION	% DAMAGE LOSS FLIGHT 133			
DATA	Full Rate		Quarter Rate	
	Uncompressed	Compressed	Uncompressed	Compressed
Compression Ratio	1·00	3·23	4·00	12·44
TAILPLANE	0	0·15	6·00	6·[illegible]0
FIN	0	0	1·20	1·20
OUTER WING	0	0·60	2·10	2·10
INNER WING	0	0	0·98	1·00
INNER WING + U/C	0	0	8·30	8·30

Fig. 9 HARDWARE CONFIGURATION

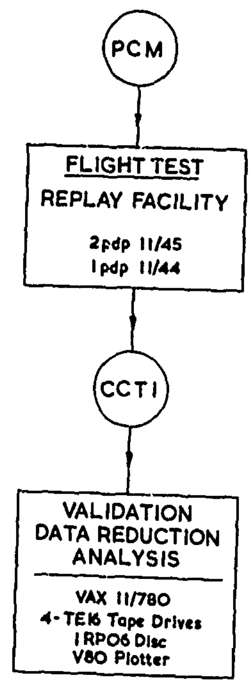


Fig 10 VAX SOFTWARE

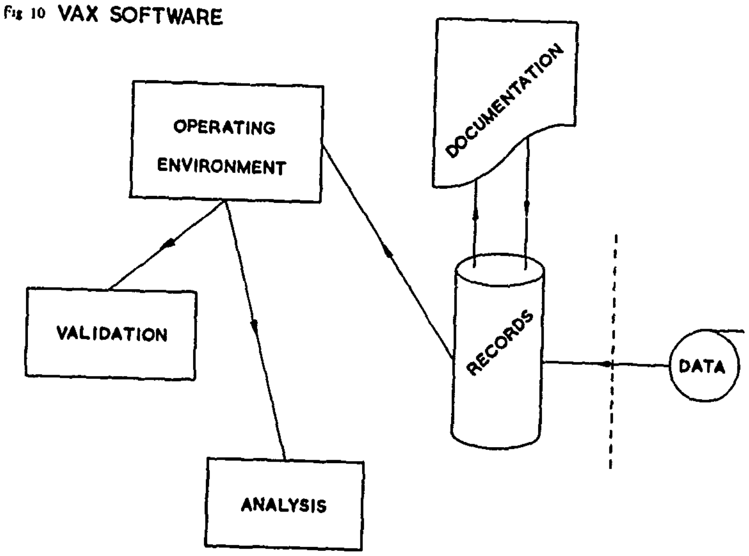
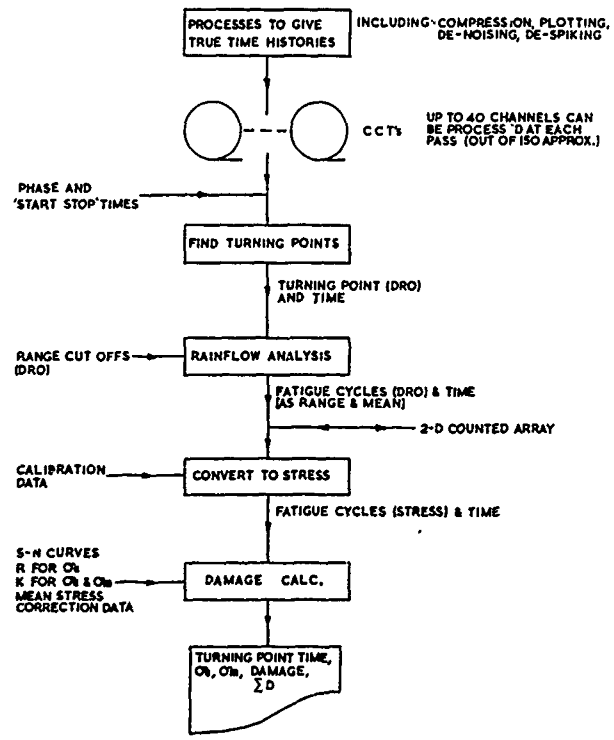


Fig. 11 PERFORMANCE
TYPICAL 8 HOUR FLIGHT

	FULL SAMPLING	1/4 SAMPLING	VAX 11/780 MILL	TAPE DRIVES
COMPRESS	24 Tapes 2:1 Ratio 12 Tapes 8 hrs.	6 Tapes 2:1 Ratio 3 Tapes 2 hrs.	85 %	3
PLOT	12 Tapes 24 hrs.	3 Tapes 6 hrs.	35%	2-3
DUPLICATE	12 Tapes 4 hrs.	3 Tapes 1 hr.	5%	2
FATIGUE	12 Tapes 5 hrs.	3 Tapes 1·25 hrs.	85%	1
MASK	4 Tapes 1·3 hrs.	1 Tape 0·3 hrs.	15%	2
	TOTAL 42·3 hrs.	TOTAL 10·6 hrs.		

Fig. 12 THE CALCULATION OF FATIGUE DAMAGE



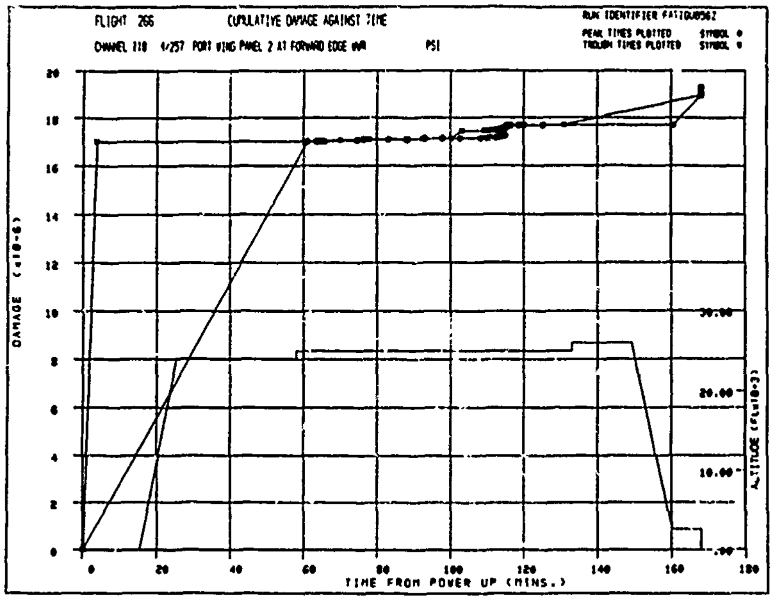

FLIGHT 266
CUMULATIVE DAMAGE AGAINST TIME
RUN IDENTIFIER FATIGU0562
CHANNEL 110 4/257 PORT WING PANEL 2 AT FORWARD EDGE WR
PSI
PEAK TIMES PLOTTED SYMBOL
TROUGH TIMES PLOTTED SYMBOL
DAMAGE (x10-6)
20
18
16
14
12
10
8
6
4
2
0
30.00
20.00
10.00
.00
ALTITUDE (Ftx10-3)
0
20
40
60
80
100
120
140
160
180
TIME FROM POWER UP (MINS.)

Fig. 13

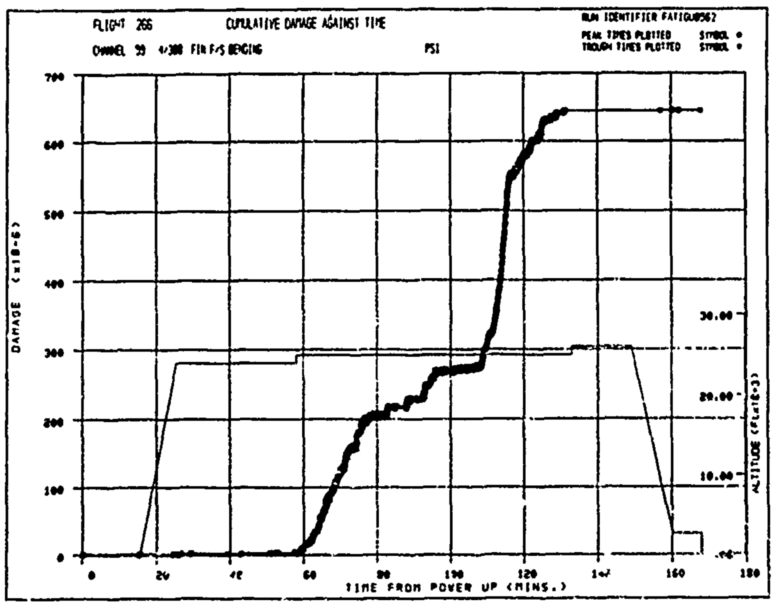

FLIGHT 266
CUMULATIVE DAMAGE AGAINST TIME
RUN IDENTIFIER FATIGU0562
CHANNEL 99 4/300 FIN F/S BENDING
PSI
PEAK TIMES PLOTTED SYMBOL
TROUGH TIMES PLOTTED SYMBOL
DAMAGE (x10-6)
700
600
500
400
300
200
100
0
30.00
20.00
10.00
.00
ALTITUDE (Ftx10-3)
0
20
40
60
80
100
120
140
160
180
TIME FROM POWER UP (MINS.)

Fig. 14

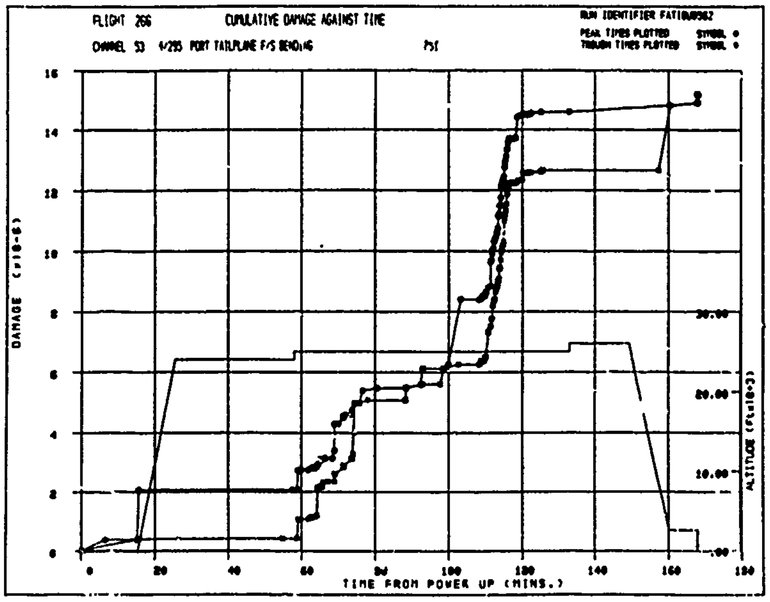

FLIGHT 266
CUMULATIVE DAMAGE AGAINST TIME
CHANNEL 53
PORT TAILPLANE F/S BENDING
RUN IDENTIFIER
PEAK TIMES PLOTTED SYMBOL
TROUGH TIMES PLOTTED SYMBOL
DAMAGE
ALTITUDE
TIME FROM POWER UP (MINS.)

Fig. 15

EXPERIENCES OBTAINED FROM SERVICE FATIGUE MONITORING EXERCISES

A P WARD
Deputy Chief Stressman
British Aerospace PLC
Warton Division
Preston PR4 1AX
England

SUMMARY

The paper reviews data obtained from empennage service loading studies performed on Jaguar and Jet Provost aircraft. The measured usage data are compared with earlier data and analysis methods are discussed.

1. INTRODUCTION

In the past service fatigue monitoring has been accomplished following procedures which were centred around the use of fatigue meter (counting accelerometer) data obtained on individual aircraft on a flight-by-flight basis, supported by additional data from the Fatigue Record Sheets for individual aircraft and individual flights, and the Statement of Operating Intent. The use of these data has previously been discussed (ref 1) and the limitations of such procedures are well known.

The paper describes two service monitoring exercises that were designed to obtain information applicable to empennage fatigue loading. The studies were performed on the Jaguar Strike A/C and the Jet Provost trainer aircraft. After general descriptions of the two programmes specific items relating to instrumentation, calibration, data analysis and results will be dealt with. A discussion of the points raised by both studies is then followed by the conclusions.

The paper is intended to highlight the main aspects of the work that might be of general interest rather than to provide a fully detailed description.

2. JAGUAR

The Jaguar is an Anglo-French single-seat, strike aircraft of conventional construction (Fig. 1). (A number of two-seat trainers were also built). Fin loading spectra were estimated from predicted sortie profiles and usage using atmospheric turbulence data and additional factors to account for pilot induced manoeuvres and aircraft damping characteristics (ref. 1).

At the end of the airframe fatigue test cracks were found in the main fin to fuselage attachment frame and, at that time, significant amounts of low level flying were predicted along with an extended fatigue life requirement. Modifications to the frame were put in hand to improve its fatigue performance in relation to these revised requirements and it was decided to embark on a "limited" service fatigue monitoring exercise to examine fin loading.

The exercise to be performed was to be accomplished at a limited cost which dictated the approach to be followed. After an initial study to examine the use of strain history measuring devices (Leigh MSR and BAe (Australia) AFDAS), or the possibility of using flight parameter data (ref. 2) it was decided to undertake a limited flight load measurement exercise.

The two strain history approaches were discarded because of the then unproven nature of these devices, which, in any case, both required a degree of load calibration on the aircraft. The parametric approach referred to in more detail in ref. 2 was discounted because of the less sophisticated mathematical response model available for the Jaguar compared with that for the Tornado.

As a number of aircraft had a Plessey EUMS Mk II digital recorder installed for engine usage monitoring purposes it was decided to base the study on the use of this device. Two GR Mk 1 single-seat aircraft were allocated to the programme and a fatigue monitoring study was devised which was based on a limited load calibration of the fin and taileron. The various aspects of this programme, illustrated on Fig. 2, are described below:

2.1 Instrumentation

The flight standard recorders used for the analysis had 5 channels allocated to structural monitoring data collection at a total sample rate of 128 s.p.s.

These five channels were allocated as follows:

- three strain gauge bridge signals at 32 sps each
- true air-speed at 2 sps
- altitude at 2 sps

The most important area of the structure was assessed to be the attachment of the centre spar of the fin to the fuselage frame (the tailerons are also mounted on this frame) Fig. 3. As a result of a study of the fatigue test results and an assessment of the structural analysis data it was decided to assign the three strain gauge channels to measurement of fin shear and torque and fuselage frame end loads at the fin centre spar pick up position. On the first aircraft a 10 Hz filter was incorporated in the recording system whereas a 40 Hz filter was incorporated on the second aircraft following further assessment of the frequencies involved.

The flight measured data were recorded on to cassettes which had a capacity of between one and three flights of data.

2.2 Load Calibration

The critical load paths were considered to be the centre spar, rear spar and frame boom. Two shear bridges were installed on each of the two spar webs close to the fuselage attachments in order to measure fin loads. The frame boom was strain gauged with eight longitudinal and four transverse gauges which were to be used to give a four-arm end load bridge on either side of the aircraft centre-line Fig. 4.

During the load calibration the best combination of spar shear bridges, (one on each spar), was chosen to yield shear force and torque. Similarly the choice of frame boom strain gauges was based on a preliminary set of fin and taileron load cases from which two fully wired 4-arm bridges were defined. These two bridges were then calibrated against fin shear and torque and the effect of taileron load was studied. As a result of this one of the two bridges was finally selected for use in service monitoring.

During load calibration the desired loads were applied in 20% increments and decrements with strain gauge responses being recorded at each stage. Prior to the measurements several exercising load cycles were applied.

The calibration analysis consisted of six phases (Table 1) and the load cases were as summarised in Table 2.

Calibration equations were established using regression techniques. Maximum reprediction errors of 1.15% and 2.25% of limit load for fin shear and torque respectively were achieved. Frame end load errors in terms of fin shear and torque were less than 0.5% of limit load. However the introduction of taileron loads degraded the accuracy somewhat, the error on fin shear reprediction increasing to 1.8% of limit load and for fin torque to 4.3%. These results are summarised on Table 3.

2.3 Electrical Calibration

The load calibration yielded relationships between load and strain gauge bridge output in millivolts. An electrical calibration was also necessary to establish the relationship between EUMS recorder output in volts and the strain gauge output in millivolts. This calibration was repeated at intervals during the flight measurement exercise in order to maintain the accuracy of the system.

2.4 Data Reduction

Following load callibration the aircraft were returned to service to fly unrestricted to typical RAF sorties. Once per week cassettes were despatched by the RAF for analysis. Each cassette was accompanied by paperwork that identified details of the flight for which data had been recorded.

Initially the cassettes were despatched to the firm responsible for the engine usage monitoring exercise for conversion to computer compatible tape (CCTs). The CCTs were then processed at Warton to yield hard copy traces of strain gauge volts for examination prior to subsequent computer analysis. During the first few months of the study a large proportion of the data was found to be corrupt. The following features, which made straight forward analysis impossible, were evident.

- random signals of opposite sign for the two fin shear bridges oscillating at high frequency
- considerable "drop-out"
- considerable "datum-shift"
- sudden datum shift
- spikes (either electrical in origin or data drop out)
- interference at 400 Hz

Fig. 5 gives examples of these problems. Various steps were taken which eventually overcame these problems to a large extent. The aircraft instrumentation systems were checked and the power units found to be faulty. Bridge integrity and channel identities were checked.

The analysis of the cassettes to produce CCTs was transferred to BAe making use of a Plessey replay unit and PDP11-55 computer prior to processing in the IBM. New Data Acquisition Units which included modifications to enhance data recovery, were fitted to the aircraft.

The time history plots of the three strain gauge signals were examined for each flight prior to subsequent processing and counts of parity errors were produced. It was initially decided that an acceptable standard would be set at a parity error count of 1 in 100 on the assumption that data quality would improve. Subsequently the standard was raised to 1 in 300 to ensure the relatively low magnitude spurious signals not filtered out during analysis had a negligible effect on the fatigue damage. This also had a significant effect in reducing the time and cost associated with analysing poor quality data.

The analysis of each acceptable CCT included tests on each data point for parity error, range limit and rate of change limit. Data failing these tests were rejected and printed out in the final output for further examination.

2.5 Analysis

Usable data obtained from the CCTs was first of all synchronised so that, by linear interpolation, signals for the centre and rear spars and the frame boom were associated with the same instant in time. These signals were then converted into loads and fatigue stresses using the calibration algorithms in association with stress/load relationships derived from the fatigue test result and the structural analysis. Stress and load time histories were produced and a rainflow analysis performed to yield stress spectra. Fatigue damage was calculated on a sortie basis and information produced to aid fleet management.

2.6 Results

The final results were based on data from 298 cassettes from which 135 flights were found to be usable. The primary fatigue loading action was that due to atmospheric turbulence at low level. Data were analysed on a month-by-month basis and a seasonal effect appeared to be present, with the highest damage rates occurring in the winter months. (Fig. 6). The fin fatigue load spectrum (Table 4) was found to have a slight bias giving more loading to port rather than starboard. Subsequent investigations identified this as a real effect due to slight trim characteristics of the aircraft concerned.

3. JET PROVOST

3.1 Background

The Jet Provost is a single engined two seat trainer aircraft which derived from the earlier Piston Provost and which was subsequently developed as the Strikemaster for use by overseas airforces (Fig. 7).

Fatigue monitoring for this aircraft was accomplished by using the fatigue meter, flying log and mission profile data. A limited amount of flight load investigations were performed many years ago. The wing, which is single spar, was strain gauged and calibrated to assess bending moment versus 'g' (normal acceleration) relationships. This was an exercise performed during development flying and the results were used, in conjunction with the fatigue meter, to assess wing fatigue life.

The empennage loading was measured for different phases of flying in 1958 (3). This study indicated that the most severe phase of flying was that associated with spinning. As a result the primary parameter used for fin and tailplane damage monitoring was "number of spins". Fig. 8 extracted from ref. 3 indicates the relative severity of different phases of flying.

This method of monitoring continued for many years with occasional theoretical checks being made against the operator's updated estimates of mission profiles to ensure that "number of spins" was still the most appropriate parameter and that atmospheric turbulence at low altitude was not as important. Unfortunately it turned out that the method used did not afford sufficient protection to the aircraft as a telephone message was received on Friday April 18th 1980 to say that a Jet Provost had made an emergency landing after the pilot had felt a "mild thump" during low level flying and had looked in his mirror to see 'the top of the fin moving laterally 12 to 18 inches" (300 to 460 mm).

On investigation the main spar booms and web of the fin (Fig. 9) were found to have failed, the fin being retained on the aircraft by the thin (0.6mm) skin attachment to the root fittings and a 0.5mm shroud. (Fig. 10). This incident led to a number of activities being necessary, one of which was a flight measurement exercise to examine empennage fatigue. This particular study is reported herein.

3.2 Formulation of Requirement

The aircraft on which the fin spar failure occurred was operated primarily in a low level navigational role whereas many other aircraft operated in mixed training roles. It was decided that two aircraft should be used for the study to investigate both types of flying.

The measurements to be made were concerned with in-flight fatigue loading of the fin and tailplane. A limited number of associated flight parameters, which were considered to provide useful data connected with sortie profile definition, were also recorded. It was thought that these latter measurements might aid future fatigue monitoring of the fleet. The loads and parameters, with their associated sampling rates, are shown on Table 5.

Aircraft altitude and speed were recorded in order to provide a comparison with similar information given in the operator's Statement of Intent. Normal acceleration was measured to provide a check against fatigue meter data and to compare with the wing spar boom strain gauge data. Such information was to be used for longer term assessments of the validity of fatigue meter data and analysis techniques. The lateral accelerometer measurements were included in order to assess whether, in the long term, a lateral acceleration counter similar to the fatigue meter might be used for fin fatigue monitoring on a fleet-wide basis.

3.3 Instrumentation

The instrumentation used was similar to that used in the Jaguar exercise with improvements wherever possible.

The data recording was accomplished using the Plessey EUMS Mk II recorder which, for this study, was capable of recording 256 samples per second on cassettes having a running time of 2½ hours.

Four-arm strain gauge bridges were used to measure fin and tailplane loading, and to obtain optimum responses, the various bridges were duplicated. The most appropriate bridges were then chosen as a result of the calibration exercise.

The four transducers used to measure altitude, airspeed and normal and lateral accelerations covered the following ranges:

Altitude (-1000 ft to 70000 ft), Airspeed (0 to 500 kt). Normal Acceleration (-5g to +12g) and Lateral Acceleration ($\pm$1g).

In addition to the above strain gauge amplifiers and data acquisition units were included in the system (Fig. 11). Following completion of the installation verification checks were performed by both Plessey and BAe engineers.

3.4 Ground Calibration

The ground calibration on each aircraft consisted of two phases, these being the structural loads calibration and the system electrical calibration.

The load calibration was achieved by applying loads in both directions to both fin and tailplane using a range of loading points representative of anticipated flight conditions. A full Skopinsky calibration (Ref. 4) was not performed.

For each loading case the strain gauge bridge responses, including duplicates, were recorded on ground monitoring equipment. From these data the choice of the optimum bridges was made and these were then wired into the aircraft system. The system was then calibrated electrically to yield the relationships between recorded signals and strain gauge bridge outputs.

In order to check that the system was functioning correctly a further loading was applied to fin and tailplane with the results being recorded on the aircraft system.

Regression techniques were used to establish the relationships between applied loads and strain gauge bridge responses.

3.5 "Shake-down" Flying

Prior to delivery of the aircraft back to the RAF a shake-down flight was performed to check that the system was functioning correctly.

3.6 Operational Checks

On delivery of the aircraft to the RAF BAe personnel visited the two stations to brief RAF personnel on monitoring procedures and system servicing. At intervals during the exercise BAe specialists performed on-site checks on the system sensitivities and datums.

3.7 Analysis

Fig. 12 illustrates the analysis procedure followed. The original intention was to obtain data on each aircraft for 6 months of flying. It very soon became apparent, because of the large number of load cycles occurring during low level flying, that analysis time and cost were considerable when compared to the original estimates. Steps were taken to reduce these by:

(i) selecting the number of flights to be analysed

(ii) selecting the parts of the flights to be analysed

In the first case this was achieved simply by making a judgement of an adequate sample size in each sortie code based on fluctuations of damage rate with increasing sample size.

In order to determine the parts of a flight to be analysed Versatec plots (hard copy) of the data on each cassette were produced from the CCT prior to further analysis. The inactive regions on the tape were identified and the remaining parts of the tape were then specified for subsequent analysis.

The data to be analysed were then examined by a checking program which searched for parity errors, and exceedances of rate of change and range limits (as in the Jaguar study). This program output data sets containing the time intervals of omitted data (rejected during the checks) for subsequent use in the analysis program.

The analysis produced a load spectrum matrix (60 x 30) of mean and alternating loads for each of the load-time histories selected for analysis (Table 6). These matrices were updated after each analysis, for each sortie code, and damage determined.

The damage per sortie code was determined

(i) by assuming damage rates for the omitted time intervals were the same as those for the analysed time intervals and

(ii) by assuming that no damage occurred during those periods identified as inactive from an examination of the Versatec plots.

Table 7 summarises information concerned with the time intervals. It can be seen that, in general, the quality of data was very good. The ratio of Actual Analysis Time/Specified Analysis Time was seen to vary between 0.86 and 0.97 for the aircraft involved mainly in low level flying, and between 0.96 and 1 for the other aircraft. The Specified Analysis time varied between 44% and 57% of the recorded Flight Time. It should be noted that this latter parameter was not measured in the flight load monitoring programme. It was the time recorded on the fatigue meter records after each flight by either the pilot. or ground crew and is recorded to the nearest 5 minutes. However, as this time is the figure used by the RAF for fleet monitoring all damage was related to it.

3.8 Results

The most damaging loading action was found to be that due to flight through turbulence at low level. Not only was the turbulence a problem but additional loading due to a 0.5 Hz dutch-roll response was also very evident. This was also identified in the 1959 study (Ref. 3). The severity of the loading was significantly more severe than previously identified. This was attributed to a number of factors.

(i) time at low level was much greater

(ii) cruise speeds at low level were between 1.5 and 2 times greater than in ref. 3 (updated engines in later Marks of aircraft)

(iii) the "low-level" flying was at a lower altitude than in ref. 3.

The time and speed at low level were compared with the figures quoted in the most recent Statement of Operating Intent. In both cases the measured data were greater than assumed.

A limited assessment of the use of lateral 'g' as a monitor of fin loading was made. A correlation coefficient of 0.981 was obtained for the low level (most damaging) flying and this became 0.970 for variable altitude flying. Poor correlation was achieved in spins, however this may have been due to the lateral 'g' sampling rate of 8 sps being inadequate to identify the higher frequency fin loads occurring in a spin.

4. DISCUSSION

Both the Jaguar and the Jet Provost studies were successful in terms of providing useful fatigue data. In both cases fin loading spectra were determined which were more damaging than the then current estimates. The analysis of time histories by the "rainflow" technique in order to generate mean and alternating load matrices identified cycles about both positive and negative mean loads as well as those about zero mean. Direct comparison between the two aircraft is therefore difficult to illustrate. If it is assumed that mean load is of second order importance then exceedance spectra of alternating load can be produced. These are shown on Fig. 13 plotted as percentage of ultimate load. The typical spectrum shown in Ref. 1 and the overall spectrum for the Jet Provost from ref. 3 are also shown for comparison. The Jet Provost spectrum is seen to be particularly severe on this basis.

The Jaguar studies commenced by making use of recording facilities and analysis processes already in existance for other purposes. This approach was not successful and modifications to the programme were necessary. Subsequently considerable useful experience was gained and the Jet Provost studies benefited accordingly.

The use of hard copy quick look plots of time histories (Versatec plots) prior to computer analysis was found to be essential in order to eliminate useless data and to reduce the amount of computer time by identifying active analysis time slices. Current studies are aimed at automating this process.

A high degree of data recovery was found to be possible in the Jet Provost programme (between 96% and 100% on one aircraft).

5. CONCLUSIONS

Two recent empennage fatigue load monitoring exercises have been described. In each case digital recording techniques were used and rainflow analysis techniques were employed to determine fatigue loading matrices. Problems were encountered in analysing the recorded data and these have been enumerated. The more recent assessment of the Jet Provost provided a high rate of data recovery and benefited from experiences gained from the Jaguar exercise. The conventional method of assessing fin load spectra was found not to provide an adequate safeguard for the later Marks of Jet Provost where significant changes in operating patterns and speeds could not be taken into account adequately by theoretical methods.

6. REFERENCES

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4. Skopinsky TH, Aiken WS, Huston WB "Calibration of strain gauge installations in aircraft structures for the measurement of flight loads", NACA Report 1178, 1954.

ACKNOWLEDGEMENTS

The assistance of colleagues at BAe and the Royal Aircraft Establishment, Farnborough in progressing both of these programmes is gratefully acknowledged. Views expressed are those of the author and do not necessarily reflect those of BAe.

TABLE 1

JAGUAR - PHASES OF CALIBRATION ANALYSIS

1. Determine suitable gauge combinations for 2 frame end load bridges.
2. Select optimum pair of shear bridges, one on each spar, and produce shear force and torque equations.
3. Determine response equations for the 2 frame end load bridges and select the most appropriate.
4. Check the accuracy of super positions for the frame end load bridge for fin and taileron loading.
5. Check reprediction accuracy for fin shear and torque in the presence of taileron loads.
6. Check the frame end load bridge response to taileron loads.

TABLE 2

JAGUAR - CALIBRATION CASES

	FIN		TAILERON				
			PORT		STARBOARD		
CASE	P	S	UP	DOWN	UP	DOWN	
1	√						
2		√					
3				√			
4			√				
5					√		
6						√	
7	√						Fin c.p. checks
8	√						
9	√						
10	√						
11	√						
12		√					
13		√					
14		√					
15		√					
16		√					
17			√		√		
18				√		√	
19	√			√		√	
20		√		√		√	
21	√		√			√	
22		√		√	√		
23	√			√	√		
24		√	√			√	

The table shows loads applied to Port (P), Starboard (S) Up, or Down on the fin and Port & Starboard Tailerons.

TABLE 3

JAGUAR - CALIBRATION EQUATIONS - REPREDICTION ERRORS

LOADING ACTION	NO TAILERON EFFECT	TAILERON EFFECT
Fin shear	1.15%	1.8%
Fin torque	2.25%	4.3%

Maximum reprediction errors as percent limit load

TABLE 4

JAGUAR - FIN LOAD SPECTRUM

The table shows counts per hour for different combinations of mean and alternating loads expressed as percentages of ultimate design load.

Alternating loads as % ultimate

	Mean Loads as % ultimate	2.6	6.5	10.4	14.3	18.2	22.1	26.0	29.9	33.9
PORT	-39.0	0.014	-	-	-	-	-	-	-	-
	-32.5	-	-	-	-	-	-	-	-	-
	-26.0	0.027	-	-	-	-	-	-	-	-
	-19.5	0.933	-	-	-	-	-	-	-	-
	-13.0	13.3	0.068	-	-	-	-	-	-	-
	-6.5	1877	18.8	1.68	0.304	0.061	0.014	0.0068	-	0.0068
	0	29740	162.3	13.3	1.602	0.250	0.041	-	-	-
STARBOARD	6.5	446	0.778	0.250	0.047	-	-	-	-	-
	13.0	30.1	0.020	-	-	-	-	-	-	-
	19.5	0.101	-	-	-	-	-	-	-	-
	26.0	0.0068	-	-	-	-	-	-	-	-
	32.5	-	-	-	-	-	-	-	-	-
	39.0	-	-	-	-	-	-	-	-	-

TABLE 5

JET PROVOST - MEASURED PARAMETERS

CHANNEL	PARAMETER	SAMPLES PER SECOND
1	Fin root shear at forward pick-up	28
2	Fin root shear at rear spar	28
3	Fin root bending moment at rear spar	28
4	Tailplane front spar bending, port	28
5	Tailplane front spar bending, starboard	28
6	Tailplane rear spar bending, port	28
7	Tailplane rear spar bending, starboard	28
8	Wing main spar lower boom strain	28
9	Aircraft altitude	4
10	Aircraft speed	2
11	Normal acceleration at aircraft cg.	8
12	Lateral acceleration at base of fin	8
13	Undercarriage selection	2

TABLE 6

JET PROVOST - ANALYSED LOAD-TIME HISTORIES

- Fin side load
- Fin root bending moment
- Fin forward attachment shear
- Port tailplane root bending moment
- Starboard tailplane root bending moment
- Port tailplane attachment loads
- Starboard tailplane attachment loads

TABLE 7

JET PROVOST - SUMMARY OF ANALYSED DATA

SORTIE CODE	LOW LEVEL NAVIGATIONAL ROLE				MIXED TRAINING ROLE			
	FLIGHTS NO	FLIGHTS TIME	ACTUAL / SPECIFIED	SPECIFIED / FLIGHT TIME	FLIGHTS NO	FLIGHTS TIME	ACTUAL / SPECIFIED	SPECIFIED / FLIGHT TIME
1	-	-	-	-	10	27	0.96	0.72
2	19	65	0.95	0.86	8	54	1.00	0.78
3	-	-	-	-	3	58	0.98	0.75
4	16	58	0.97	0.71	7	54	0.98	0.74
5	-	-	-	-	11	66	0.99	0.59
6	9	61	0.86	0.64	17	62	0.98	0.69
7	-	-	-	-	5	63	0.99	0.69
8	-	-	-	-	4	51	1.00	0.44
9	-	-	-	-	1	20	0.96	1.58

Time : Minutes
Actual : Actual time analysed in each flight
Specified : Specified time to be analysed in each flight
Flight Time : Flight duration recorded on fatigue record sheets

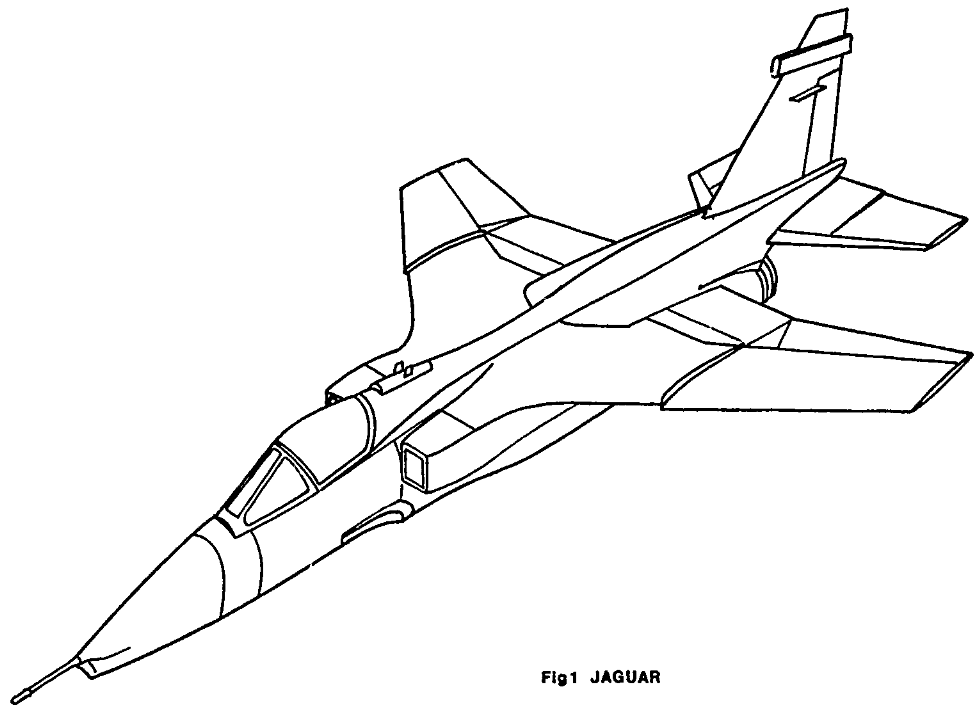


Fig1 JAGUAR

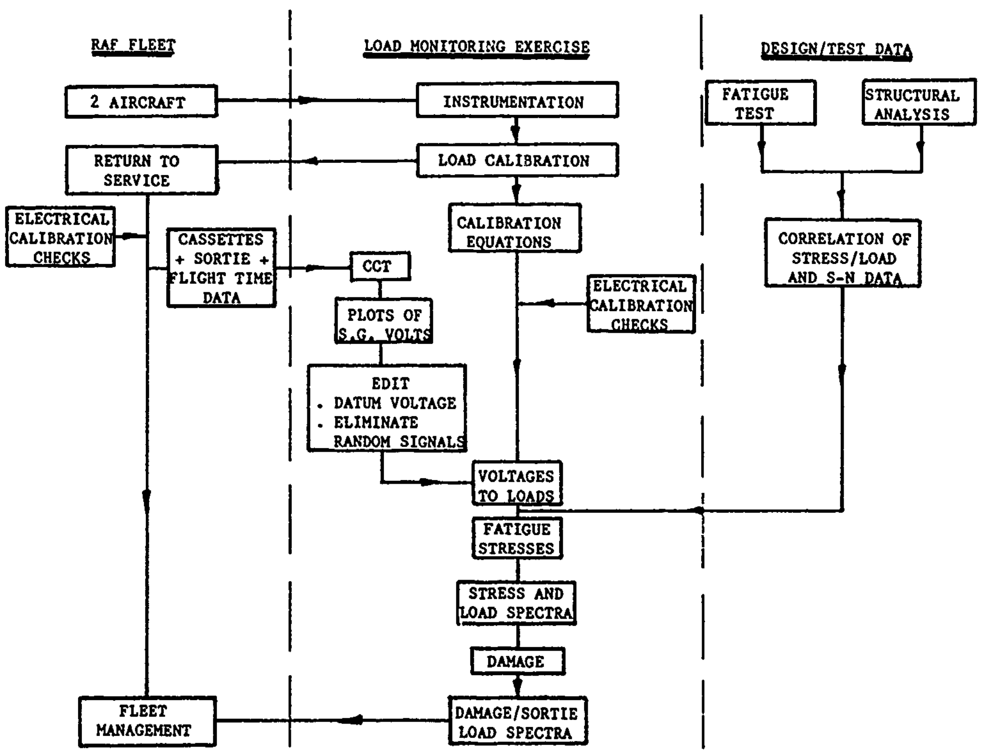


FIG. 2 Summary of Jaguar Load Monitoring Study

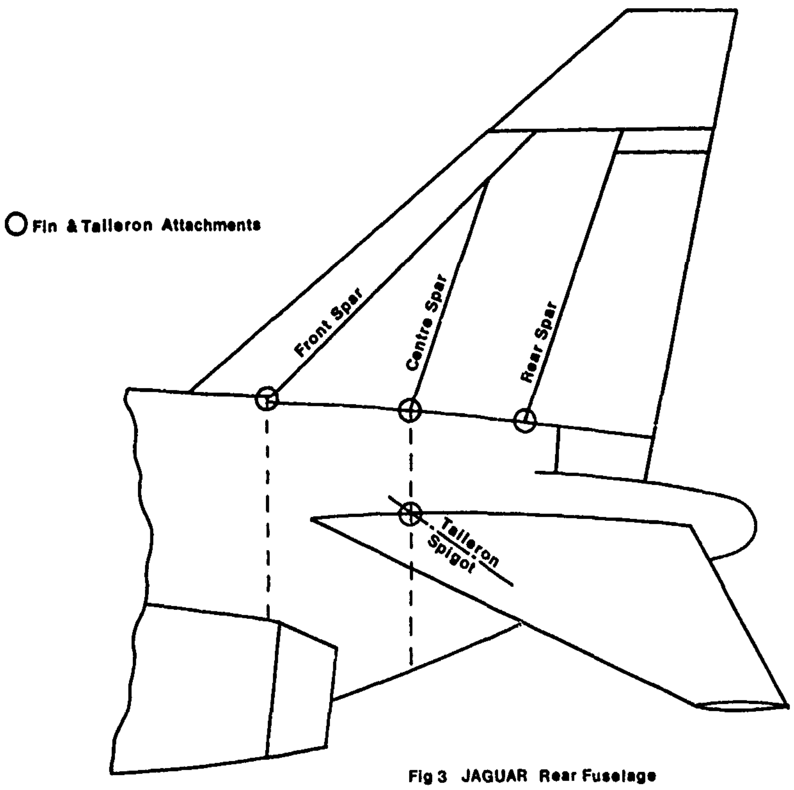


Fig 3 JAGUAR Rear Fuselage

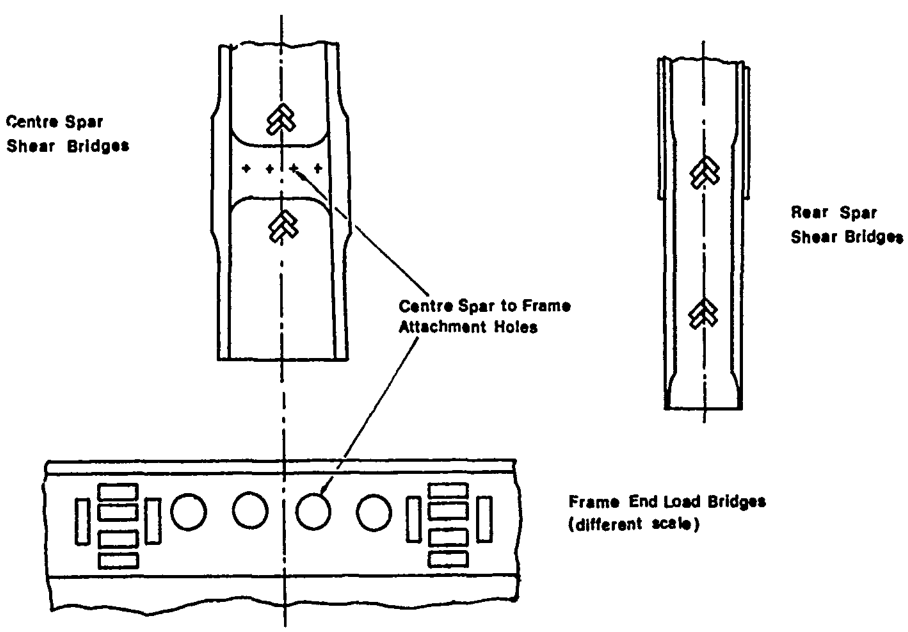


Fig 4 Strain Gauge Locations

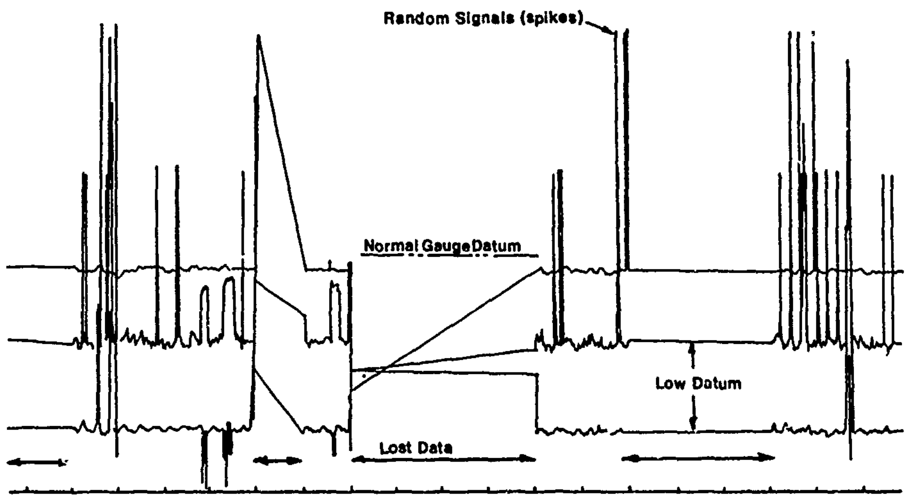


Fig 5 Examples of Corrupt Data

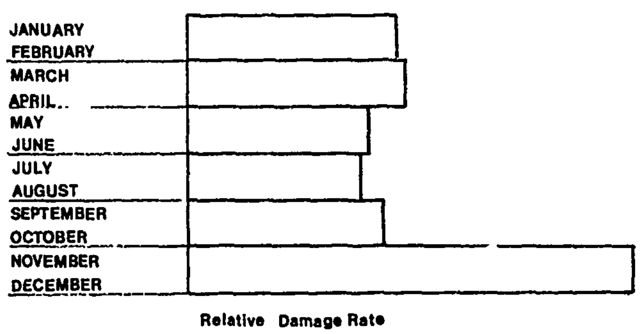


Fig6 JAGUAR Possible Seasonal Effect on Fin Damage

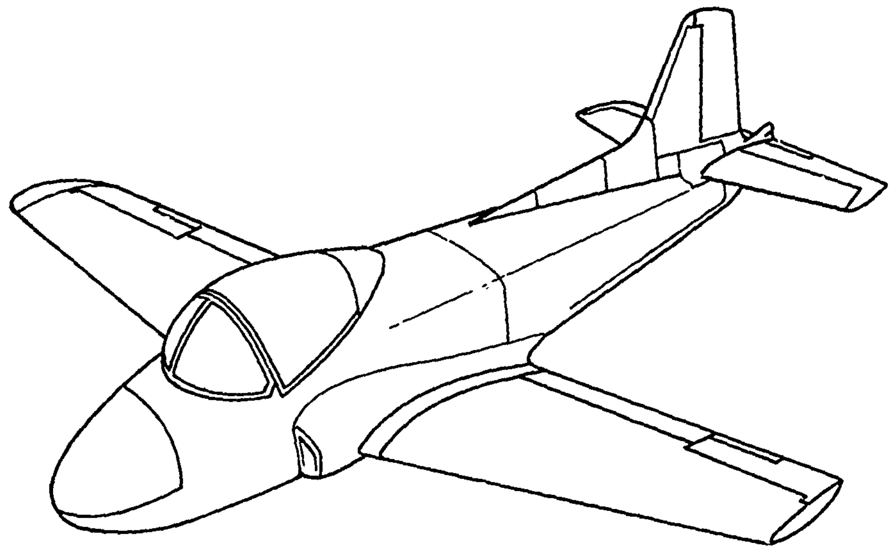


Fig 7 JET PROVOST

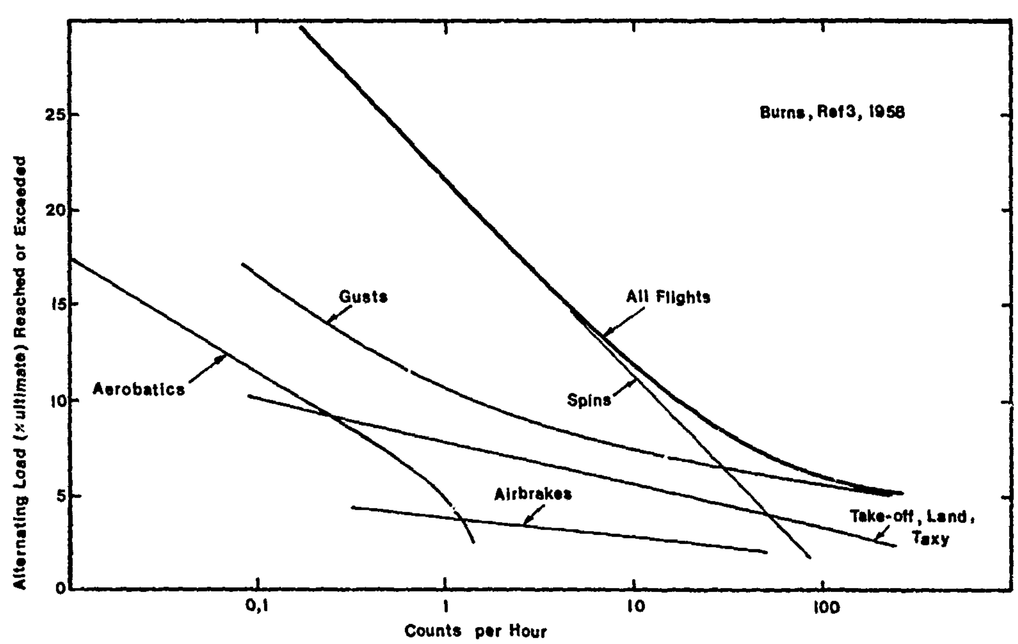


Fig 8 Fin Load Spectra

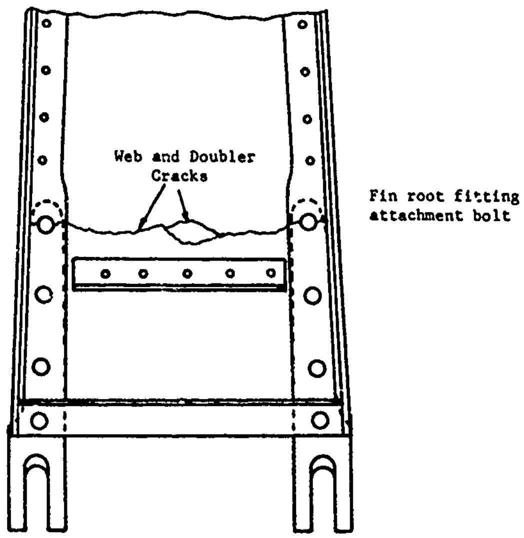


FIG. 9 JET PROVOST - Failed Fin main spar (booms, web and doubler cracked)

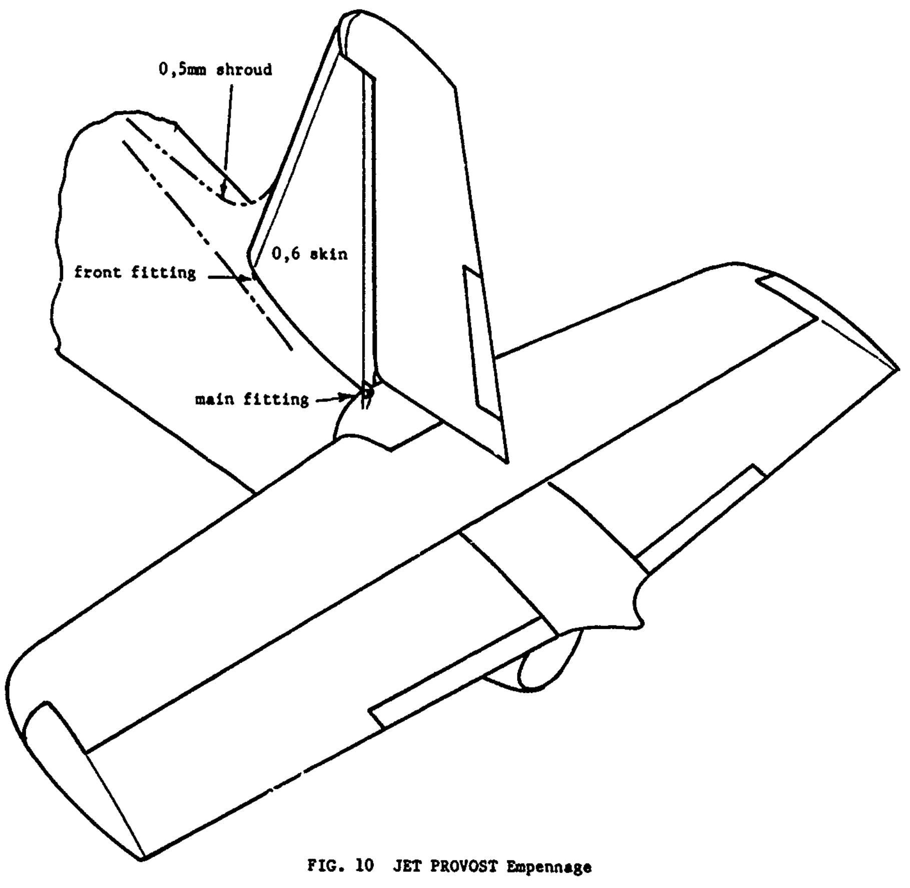


FIG. 10 JET PROVOST Empennage

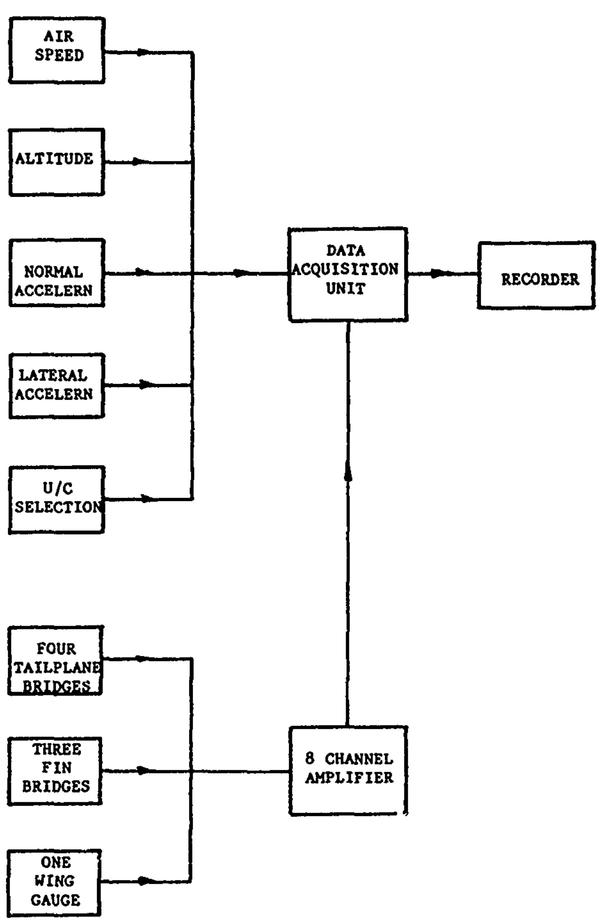


FIG. 11 JET PROVOST - Block Diagram of Strain Gauge Instrumentation, Transducers and Associated Recording Equipment

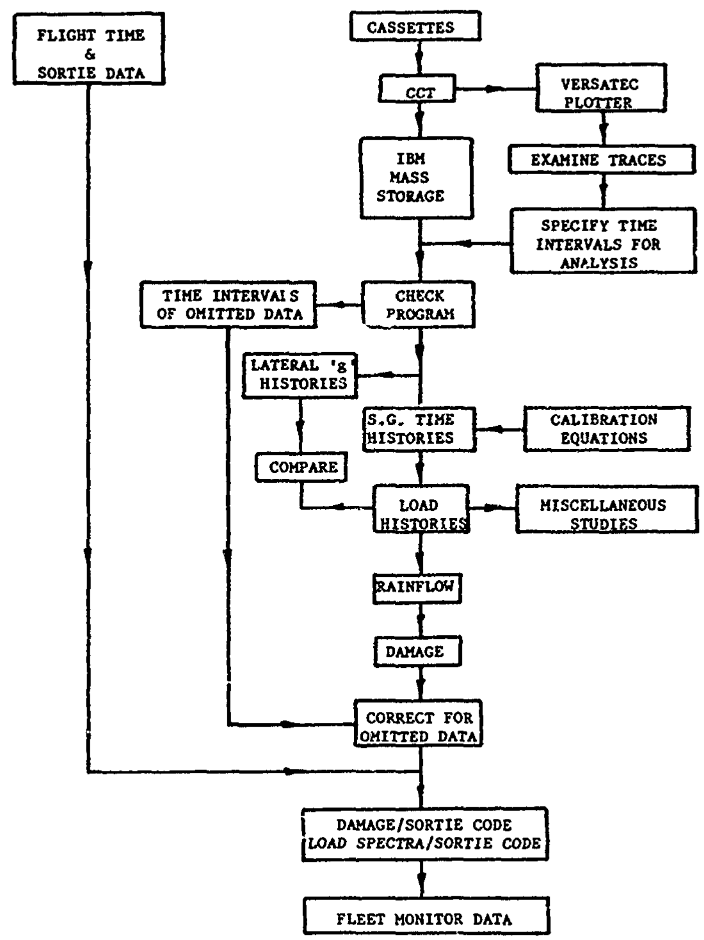


FIG. 12 JET PROVOST - Data Analysis

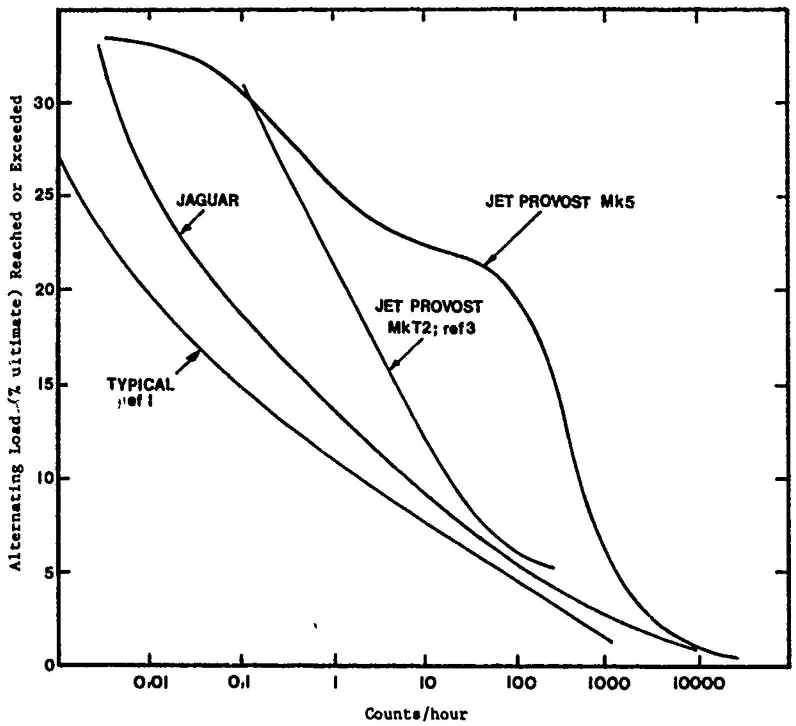


FIG. 13 Fin Load Spectra

CF-5 VERTICAL STABILIZER FLIGHT LOAD SURVEY

by

N. Nguyen Trong
Military Support, Technical Department
Canadair Limited
P.O. Box 6087, Montreal, P.Q.
Canada, H3C 3G9

SUMMARY

The CF-5 Vertical Stabilizer has long been recognized as a critical structural item. In order to arrive at a reliable life prediction for this particular structure, a flight load survey was carried out to determine the load spectra.

Specific missions were flown with an instrumented aircraft and the resulting load spectra were factored to represent the reported mission profiles for squadron aircraft. Subsequently, it was decided to monitor a percentage of the fleet during routine operation and derive more representative spectra.

This paper compares the results of the flight load survey and the load monitoring program and their effect on the fatigue life prediction and damage tolerance analysis results.

It concludes that a short flight test program gives valid results which are useful for immediate use, and that the fleet wide load monitoring program consolidates and verifies the findings.

1. INTRODUCTION:

Northrop fatigue test on a F-5E full-scale original design Vertical Stabilizer showed cracks in the rear fuselage formers and stabilizer-to-fuselage attachment angle. More importantly, this cracking was followed soon after by a fatigue crack at the side skin root radius, (Figure 1). This skin is considered most structurally critical as its failure will lead to the loss of the Vertical Tail. As a result, Northrop subsequently changed the design of the F-5E Vertical Stabilizer. Inspection of the CF-5 aircraft revealed a high percentage of the fleet having cracks in the rear formers similar to those found in the F-5E fatigue test. Since the CF-5 Vertical Stabilizer is identical to the one used in the F5E test, this prompted concern for its fatigue and damage tolerance lives.

Consequently, Canadair was asked by the Canadian Forces to establish the safe life, damage tolerance characteristics and inspection intervals for the CF-5 Vertical Stabilizer. This led to the need to obtain the CF-5 Vertical Stabilizer actual load spectrum by means of a flight load survey program. This was considered necessary as the load spectra previously used in Northrop analytical studies had been based on design values for the frequencies of encountering dynamic yaw and rolling pull-out conditions. Two survey programs were carried out as follows:

a) A short-term limited flight test program with an instrumented aircraft. This provided an indication of the overall flight load spectrum.

b) A long-term load monitoring program on a percentage of the fleet using Mechanical Strain Recorders (MSR). This program yielded a more representative load spectrum than that obtained from the short-term program.

2.0 SHORT-TERM FLIGHT TEST PROGRAM:

2.1 Instrumentation:

The instrumentation system used in the flight test was entirely on board the aircraft and consisted of sensors for both flight and structural parameters. These sensors included strain gauges, skin surface temperature gauges and lateral accelerometer. The locations of the strain gauges on the port and starboard Vertical Tail side skins are shown in Figures 2 and 3 respectively. All the gauges were used in the ground calibration and gauges 6, and [illegible] each on both the port and starboard sides) were selected to produce the Vertical Tail lateral bending moment.

2.2 Ground Calibrations:

The ground calibrations of the strain gauges were done by Canadair with lateral load applied to the Vertical Stabilizerby means of a hydraulic jack. The applied load was measured by a load cell between the jack and the tail surface. Readings were taken directly from the strain gauges, without going through the aircraft instrumentation system. A detailed description of the calibration is contained in Ref. 2. Four gauges 6P, 6S, 7P and 7S (Figures 2 and 3) were essentially linear with respect to the lateral bending moment M_x and were not affected by torsion. As a result, these four gauges were selected to measure in-flight lateral bending moments on the Vertical Stabilizer at WL17.58". All four gauges were sufficiently accurate throughout their range that they could be used to cross check and back up each other. Furthermore,the responses of gauges 1P and 1S were also recorded to provide an instrumentation redundancy.

2.3 Preliminary Flight Tests:

To ensure proper operation of the instrumentation and build up confidence in the load measurement procedure, a series of preliminary flight tests were carried out. The manoeuvre performed was a rudder reverse manoeuvre. The aircraft was stabilized in a wings level sideslip with full right rudder deflection (i.e., deflected to its hinge-moment limit). The rudder was then reversed at the maximum possible rate and held at full left deflection until the aircraft stabilized. Wings were held level throughout with the application of aileron. Lateral bending moment was measured during this manoeuvre at various altitudes and speeds. The results were then compared to Northrop flight test data from Ref. 3. Good agreement was obtained and reported in Ref. 4. This confirmed that the aircraft is representative of standard aircraft for load measurement purposes.

2.4 Flight Loads Survey:

Twelve flights were executed with mission segments representative of squadron usage of the CF-5 aircraft. To provide a good coverage of possible conditions, different external store configurations, different techniques of rudder manipulation and different pilots were used. A summary of flights is given in Table 1. The pilots were instructed to maximize, rather than minimize, the magnitude and occurrence rate of sideslips by using rudder and abrupt aileron control input freely.

TABLE 1 - LOADS SURVEY FLIGHTS

FLIGHT		EXTERNAL
NUMBER	MISSIONS	CONFIGURATIONS
F1 F2 F3 F4	Air Combat Training	No external stores
F5 F6	Handling and Formation	Centerline tanks
F7	Low level navigation, reconnaissance and tactical	Centerline tanks
F8	Low level navigation, reconnaissance and tactical	WS 85 tanks PLUS SUU-20
F9 F10 F11 F12	Low level navigation and range weapons deliveries	WS 85 tanks PLUS SUU-20

No attempt was made to duplicate individual squadron mission profiles. Instead, a "building block" approach was used. For this, mission phases such as take-off, aerobatics, etc , often common to different squadron mission profiles, were isolated in each flight and flight test data was obtained for each phase. Complete missions could then be built from the appropriate mission phases. Table 2 summarizes the mission phases for which flight test data was obtained. Table 3 shows a typical mission profile build-up from these mission phases for a particular CF-5 squadron.

TABLE 2 - CF-5 Mission Phase Description

Mission Phase No.	Description
1	Taxi
2	Take-Off
3	Non-manoeuvring (included: acceleration, climb, cruise, handling stalls, recovery, visual landing patterns, TACAN, radar square, radar base, radar final, patterns "8" and "O" RTB IFR, RTB VFR, relights, refueling pattern, basic IF)
4	Handling Aerobatics (included: loop, roll, cuban eight, clover leaf, unusual attitudes, 135° slice, reversal turns, reversal pull-up, extension and confidence manoeuvres, firing pass uphill and down hill, instrument aeros, mach run)
5	Formation Handling (included: formation exercises, breaks, rejoins, trail line astern, wing position)
6	Air Combat Manoeuvres (included: attacker, defender)
7	Weapons Patterns, Low Angle
8	Weapons Patterns, High Angle
9	Landing
10	Low Level Navigation (Speed: below 400 KIAS)
11	Low Level Navigation (Speed: 400 to 450 KIAS)
12	Low Level Navigation (Speed: above 450 KIAS)
13	Route Reconnaissance

TABLE 3 - TYPICAL CF-5 MISSION PROFILE

MISSION	(A)	(B)	(C)	(D)	(E)	(F)	(G)	(H)	(J)	(K)
Mission Phase	Min.	Min.	Min.	Min.	Min.	Min.	Min.	Min.	Min.	Min.
1	7.0*	7.0	7.0	12.0	7.0	7.0	12.0	7.0	8.5	7.0
2	(1)**	(1)	(1)	(1)	(1)	(1)	(1)	(1)	(1)	(1)
3	40.5	3.0	63.5	35.0	54.5	61.5	12.5	39.75	35.25	76.75
4	3.75								0.5	
5										
6								12.0		
7		1.0		9.0			22.0			
8		1.0		9.0			22.75			
9	(2)	(1)	(1)	(1)	(2)	(1)	(1)	(1)	(1)	(1)
10		12.5		3.0						
11		43.75		9.0						
12										
13										
Total Time	65.5	71.0	73.25	80.25	74.5	69.75	72.0	60.0	47.0	85.0
Monthly Flight Rate (MFR)	3.5%	25.7%	5.0%	4.0%	5.0%	0.0%	19.8%	15.5%	18.0%	3.5%

NOTE: * The time is expressed in minute and decimal.
** The data enclosed between brackets is the number of events.

2.5 Data Reduction:

A total of 10 hours of flight data was collected from the flight tests. To establish the bending moment spectrum at the root of the Vertical Stabilizer, the bending moment magnitude and occurrences had to be extracted from the data obtained from the flight tests. For this purpose, the "Peak Between Zero"counting method was used. Under this method, the bending moments were segregated into two types - the positive and negative bending moments. A fatigue cycle was counted between two changes of the bending moment sign. The value assigned to this cycle was the maximum or the minimum peak value recorded in the cycle. Figure 4 illustrates the method.

The duration of each mission was "standardized" to one hour for ease of computing. The bending moment occurrences obtained by the "Peaks Between Zero" counting method for the "standardized" mission phases in a particular mission were summed up to yield the bending moment spectrum for each one hour mission. Knowing the Monthly Flight Rate (MFR) of each particular mission in each squadron (Table 3), the Vertical Stabilizer bending moment spectrum for each squadron was established for a flying period of 1,000 hours. Table 4 provides this spectrum for Squadron 'A' as the number of occurrences and cumulative occurrences versus Lateral Bending Moment Mx.

TABLE 4 - CF-5 VERTICAL STABILIZER LATERAL BENDING MOMENT M_x SPECTRUM OBTAINED FROM SHORT TERM FLIGHT TEST PROGRAM (SQUADRON 'A').

M_x AT WL 17.58 (lb-ins)	M_x AT WL 17.58 (N-m)	OCCURRENCES PER 1000 HRS	CUMULATIVE EXCEEDANCES PER 1000 HRS
-250,000	(-28,246)	80	80
-230,000	(-25,987)	0	80
-210,000	(-23,727)	248	328
-190,000	(-21,467)	0	328
-170,000	(-19,208)	475	803
-150,000	(-16,948)	216	1,019
-130,000	(-14,688)	456	1,475
-110,000	(-12,428)	590	2,065
- 90,000	(-10,169)	1,115	3,180
- 70,000	(- 7,909)	3,681	6,861
- 50,000	(- 5,649)	12,516	19,377
- 30,000	(- 3,390)	108,753	128,130
- 10,000	(- 1,130)	1,027,835	1,155,965
10,000	(1,130)	397,225	1,161,403
30,000	(3,390)	661,109	764,178
50,000	(5,649)	81,312	103,069
70,000	(7,909)	11,014	21,757
90,000	(10,169)	4,563	10,743
110,000	(12,428)	2,730	6,170
130,000	(14,688)	1,129	3,440
150,000	(16,948)	1,015	2,311
170,000	(19,208)	614	1,296
190,000	(21,467)	462	682
210,000	(23,727)	240	320
230,000	(25,987)	80	80
250,000	(28,246)	0	0

Figure 5 shows the plot of this spectrum.

3.0 LONG-TERM FLEET LOAD MONITORING PROGRAM:

To obtain more statistically valid data, a fleet load monitoring program was carried out for three years. About 20% of the fleet aircraft were instrumented and monitored during this period of time.

3.1 Instrumentation:

Each monitored aircraft in this program was fitted with a Mechanical Strain Recorder (MSR) as shown in Figure 6. The MSR, (Fig. 7), manufactured by Leigh Instruments Limited, records strain mechanically and is independent of aircraft systems for power or data inputs. It includes two arms which are bonded to the aircraft structure; one drives a stylus and the other holds a cassette of metal tape. Strain on the structure produces a relative deflection of the two arms, causing the stylus to etch a trace of strain history of the structure. The strain motion is also used to mechanically advance the cassette tape so that strain cycles are recorded in sequence, but not on a regular time base. No electrical or other power source is required for MSR operation. The MSR requires no maintenance, except for periodic tape changes, and therefore is attractive for long term loads monitoring. It has already found application on the USAF F-16, A-37, and A-10 fleet; and on F-5 aircraft, including CF-5s, for monitoring wing loads in a multi-national service life extension study.

3.2 Ground Calibration:

During the calibration of the electric strain gauges used in the short-term flight test program (Ref. Section 2.2), a MSR was also installed on the test aircraft and a correlation between the MSR and the electrical strain gauges was established on the basis of strain measurement. A direct relationship between the MSR micro strain and the Vertical Stabilizer lateral bending moment was obtained as follows:

$$BM = 230.75\ MSR - 1030$$

where BM = lateral bending moment in lbs at WL 17.58
and MSR = measured strain in microstrain

3.3 Preliminary Flight Tests:

Flight tests were carried out to demonstrate the structural integrity of the MSR installation and to determine its effect on aircraft flying qualities. A series of steady state and dynamic yaw and roll manoeuvres was flown with and without the MSR and cover installed to identify changes in aircraft response.

It was anticipated that any effect on aircraft flying qualities, due to the location of the MSR at the base of the Vertical Stabilizer, would be evident in changes to the aircraft's Dutch roll characteristics, available rudder travel (rudder travel is normally hinge-moment limited) and/or steady sideslip characteristics. Comparison of data obtained with and without the MSR indicated that these characteristics were essentially unchanged except at 5,000 feet and 0.92 Mach where increased right rudder travel indicated a reduction in rudder hinge moment (probably due to shock effects) and an increased ratio of sideslip angle to rudder deflection with right rudder inputs indicating an increase in the coefficient of yawing moment due to rudder deflection. It was not anticipated that these effects would noticeably alter the aircraft's flying qualities. Additional details of the flight test results and analysis are presented in Annex B of Ref. 4.

3.4 Flight Load Monitoring:

The load monitoring of the Vertical Stabilizer with the MSR unit was continuously carried out for a period of about 3 years. All monitored aircraft were to perform normal flight operations. No attempt was made to maximize or minimize the tail loads at any time. The MSR cassettes where the strain was recorded were changed periodically to avoid tape run-out.

3.5 Data Reduction:

Each used cassette was sent to Leigh Instruments for processing and the microstrain recordings read. Using the transfer function described in Section 3.2 the lateral bending moment on the Vertical Stabilizer could be obtained. The "Peak Between Zero" counting method was also used to obtain the number of occurrences at a particular bending moment level. The Vertical Stabilizer lateral bending moment spectrum was established by adding up all the occurences counted at each bending moment level from all the monitored aircraft in each particular squadron. A 1,000 flying hour basis was used to produce the spectrum.

In order to determine the validity of the data used for the spectra, a statistical test is performed. The formula (Ref. 8),

$$n = \frac{(z_{\alpha/2})^2 \sigma^2}{e^2}$$

where n = number of hours
e = error = 0.1 x mean
$z_{\alpha/2}$ = confidence level (1.96 for 95%)

σ = standard deviation

is used to calculate the hours required to be 95% confident that the error in the mean is less than 10%. The standard deviation is calculated by dividing the difference between the maximum and minimum counts per hour by 6. This method is described on Page 264 of Ref. 8) and is based on the fact that the mean plus or minus 3 standard deviations would include almost 100% of the potential observations. If the total number of hours of data that has been collected is greater than the required hours then enough data has been processed.

A total of 1273 hours of data were collected for squadron 'A'. However, about 13% of this data was not acceptable and was rejected before the reduction of data.

3.6 Results of the Flight Load Monitoring:

Table 5 provides the Vertical Stabilizer lateral bending moment spectrum obtained from the long-term load monitoring program. Figure 8 shows the plot of this spectrum. More details of the flight load monitoring program can be found in Ref. 7.

TABLE 5 - CF-5 VERTICAL STABILIZER LATERAL BENDING MOMENT M_x SPECTRUM OBTAINED FROM THE LONG-TERM FLIGHT LOAD MONITORING PROGRAM (SQUADRON 'A')

M_x AT WL 17.58		OCCURRENCES PER 1000 HRS	CUMULATIVE EXCEEDANCES PER 1000 HRS
(lb-ins)	(N-m)		
-250,000	(-28,246)	37	37
-230,000	(-25,987)	21	58
-210,000	(-23,727)	66	124
-190,000	(-21,467)	98	222
-170,000	(-19,208)	236	458
-150,000	(-16,948)	647	1,105
-130,000	(-14,688)	1,753	2,858
-110,000	(-12,428)	4,020	6,878
- 90,000	(-10,169)	7,734	14,612
- 70,000	(- 7,909)	9,318	23,930
- 50,000	(- 5,649)	5,935	29,865
- 30,000	(- 3,390)	2,329	32,194
- 10,000	(- 1,130)	646	32,840
10,000	(1,130)	5,251	32,812
30,000	(3,390)	11,156	27,561
50,000	(5,649)	8,744	16,405
70,000	(7,909)	4,659	7,661
90,000	(10,169)	1,970	3,002
110,000	(12,428)	653	1,032
130,000	(14,688)	214	379
150,000	(16,948)	83	165
170,000	(19,208)	43	82
190,000	(21,467)	15	39
210,000	(23,727)	13	24
230,000	(25,987)	4	11
250,000	(28,246)	7	7

4.0 COMPARISON OF RESULTS:

4.1 Comparison of Load Spectrum:

Figure 9 shows the two Vertical Stabilizer Bending Moment spectra obtained from the short and long-term load monitoring programs. The spectrum obtained from the short-term flight test is more severe.

4.2 Comparison of Fatigue and Damage Tolerance Analysis and Results:

Analyses were carried out to determine the fatigue and damage tolerance lives for the CF-5 Vertical Stabilizer based on the two spectra shown in Figure 9. Table 6 provides the results of these analyses.

TABLE 6 - COMPARISON OF RESULTS

	Based on Short Term Load Monitoring Spectrum	Based on Long Term Load Monitoring Spectrum	Difference
FATIGUE LIFE (HRS) INCLUDING A SCATTER FACTOR OF 4	2000 (Ref. 5)	3420	-42%
DAMAGE TOLERANCE LIFE (HRS)*	2600 (Ref. 5)	2660 (Ref. 6)	-2%

*Assuming an initial flaw of 0.050" as per Ref. 1.

5.0 DISCUSSION AND CONCLUSION:

From the comparison of results (Section 4) it is seen that the short-term flight test led to more severe load spectrum for the Vertical Stabilizer lateral bending moment. This resulted in more conservative fatigue and damage life estimates for the side skin of the Vertical Tail. This condition is somewhat expected since the pilots for short-term flight test program were instructed to exaggerate the rudder manipulation while no special instructions were given to pilots in the long-term load monitoring program.

Also, the data available for the short-term program is very limited compared to that of the long-term program (10 hrs vs 1128 hrs of data). Therefore the statistical validity of the resultant load spectrum is restricted.

However, the short-term program had the merit of being less time consuming (2 weeks vs 3 years) as well as inexpensive. The time factor was quite important in this case as a quick and relatively realistic estimate for the Vertical Stabilizer fatigue life was needed to ensure the structural integrity of the aircraft during the more long-term flight load survey.

In conclusion, the 2 different methods of load measurement used on the CF-5 Vertical Stabilizer produced quite compatible load spectra. The long-term load monitoring method is obviously preferable since a more realistic and statistically valid spectrum is obtained. However, a short-term flight test with carefully selected flight phases and accurate mission profiles can also lead to very acceptable results within limited time and cost.

6.0 REFERENCES:

1. MIL-83444 - Military Specification for Airplane Damage Tolerance Requirements, 2 July 1979.
2. Canadair Report RAS-219-102, CF-5 Vertical Stabilizer Flight Load Ground Calibration, December 1977.
3. Northrop Report NOR-66-11, CF-5 Final Loads Flight Data for Initial and Final Phase Tests, 1 December 1966, revised 1 September 1967.
4. Canadian Forces AETE Project Report 75/49 - CF-5 Vertical Stabilizer Flight Loads Survey, 4 July 1978.
5. Canadair Report RAS-219-103 - CF-5 Vertical Stabilizer Fatigue Analysis, August 1978.
6. Canadair Report RAS-219-111 Damage Tolerance Analysis of a cracked CF-5 Vertical Stabilizer Main Skin - March 1983.
7. Canadair Memorandum MAS-219-148 Vertical Stabilizer Bending Moment Spectrum, April 1983.
8. Lincoln L. Chao, "Statistics for Management" 1st Edition, Brooks, Cole Publishing Company, Monterey, California 93940, 1980, page 263.

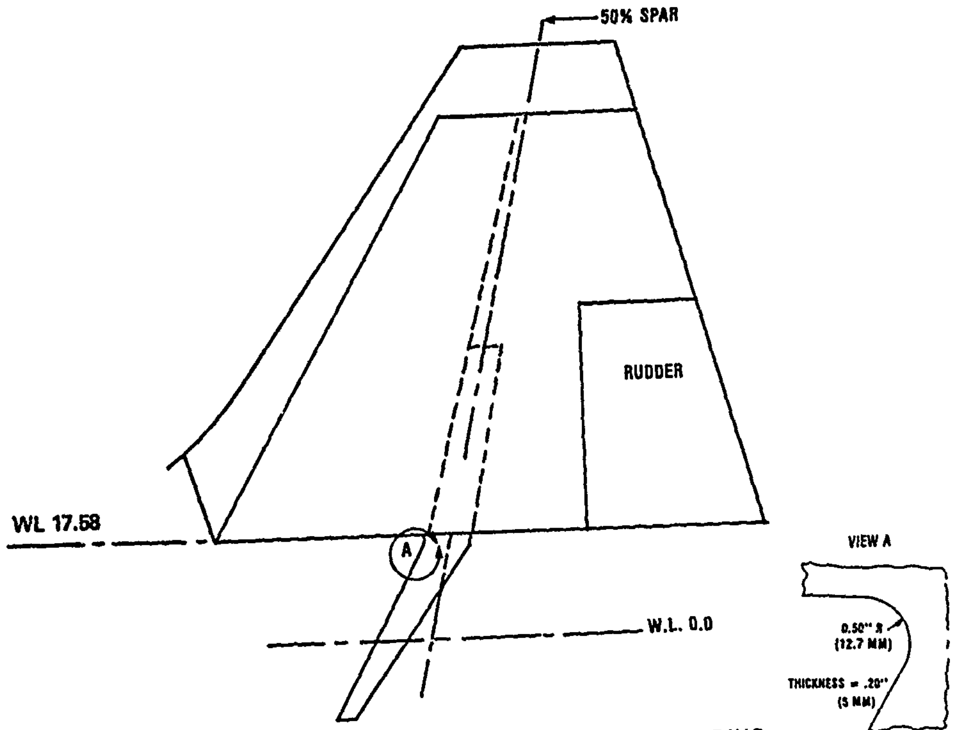


FIGURE 1 — CF-5 VERTICAL STABILIZER CRITICAL RADIUS

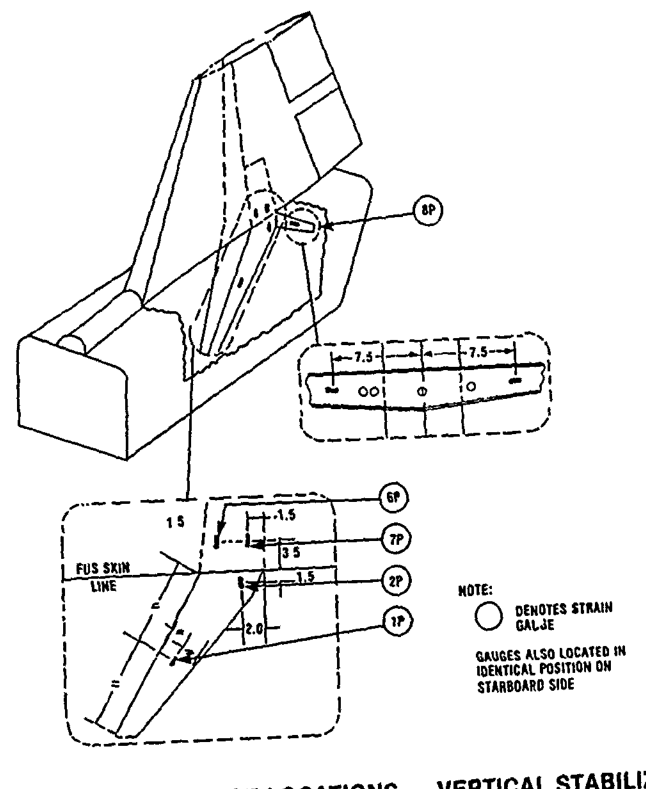


FIGURE 2 — STRAIN GAUGE LOCATIONS — VERTICAL STABILIZER
PORT SIDE

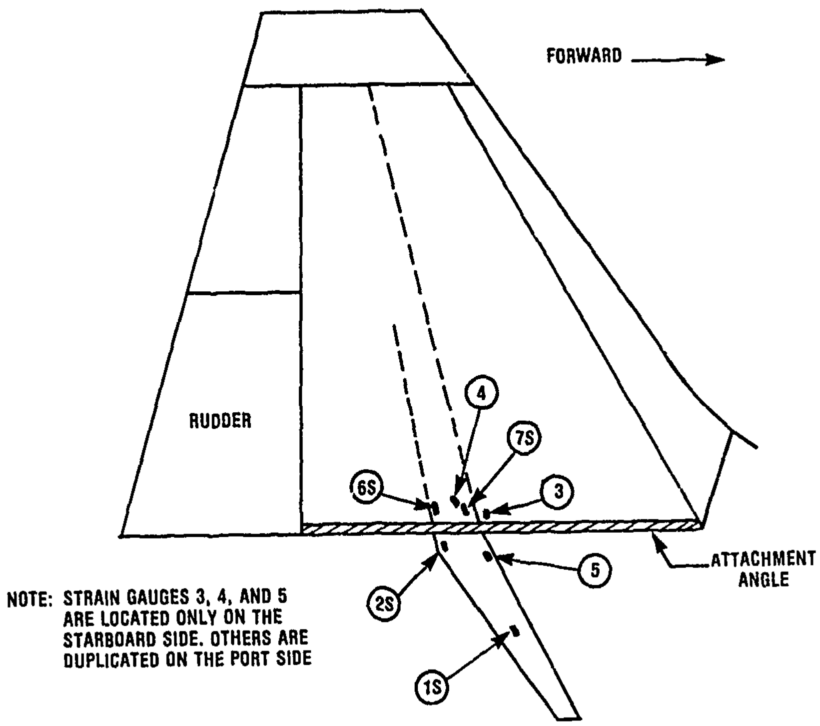


FIGURE 3 — STRAIN GAUGE LOCATIONS — VERTICAL STABILIZER STARBOARD SIDE

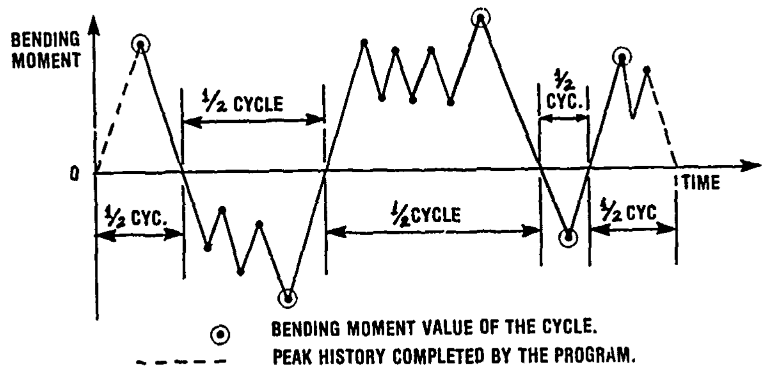


FIGURE 4 — PEAK BETWEEN ZERO COUNTING METHOD

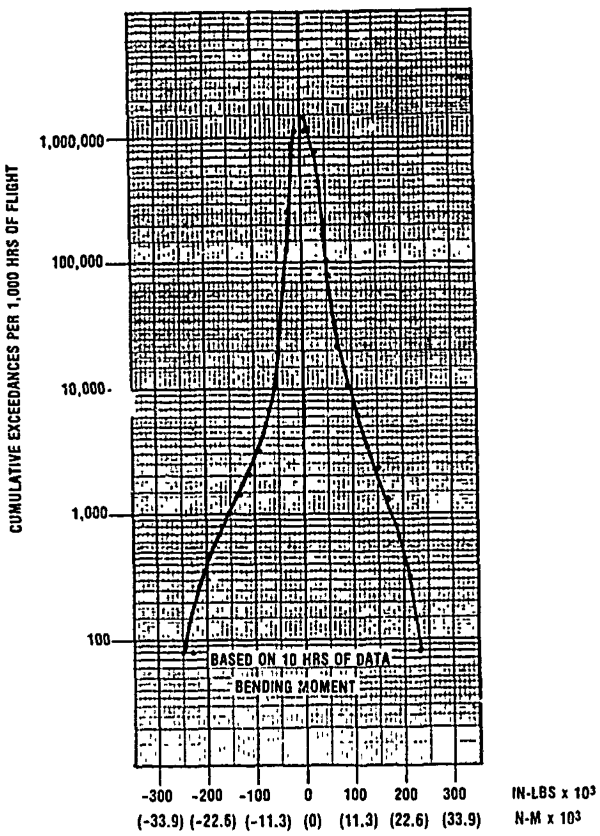


FIGURE 5 — CF-5 VERTICAL STABILIZER LATERAL BENDING MOMENT SPECTRUM OBTAINED FROM THE SHORT TERM FLIGHT LOAD SURVEY

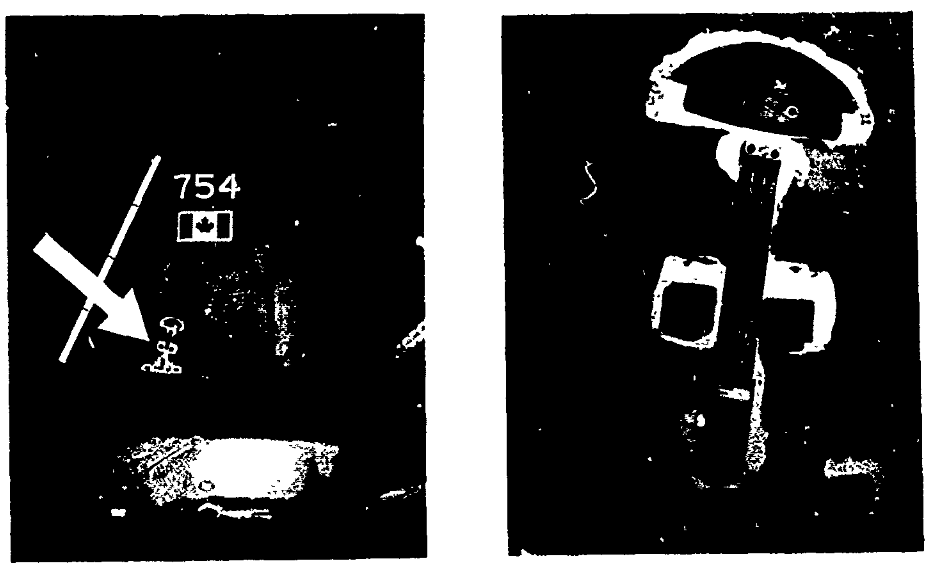


FIGURE 6 — **MSR Installation**

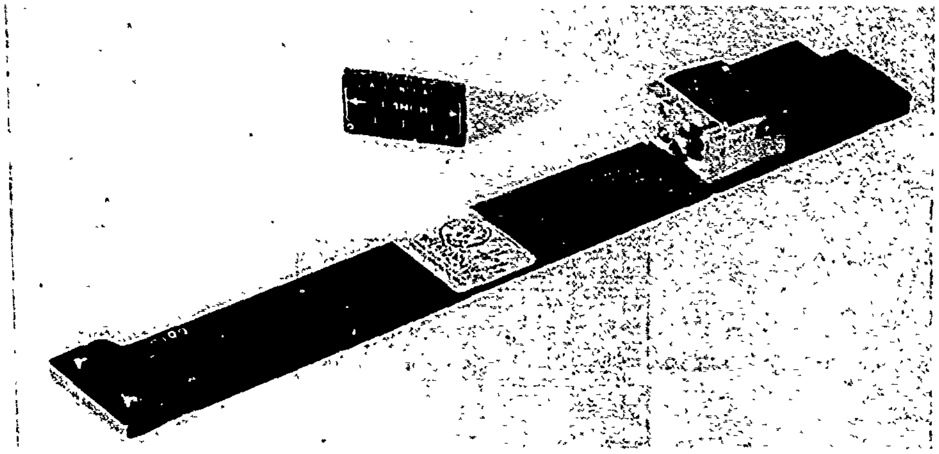


FIGURE 7 — LEIGH INSTRUMENTS MECHANICAL STRAIN RECORDER

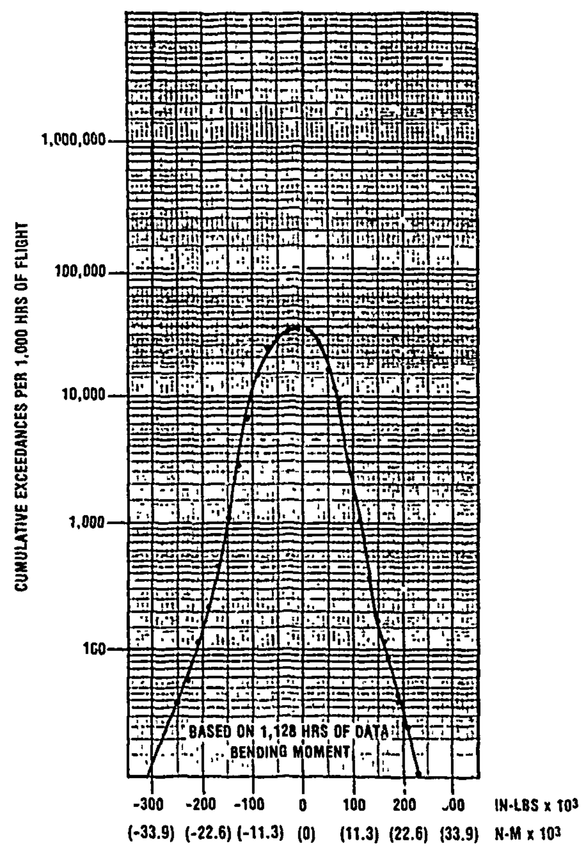


FIGURE 8 — CF-5 VERTICAL STABILIZER LATERAL BENDING MOMENT SPECTRUM OBTAINED FROM THE LONG TERM LOAD SURVEY MONITORING PROGRAM

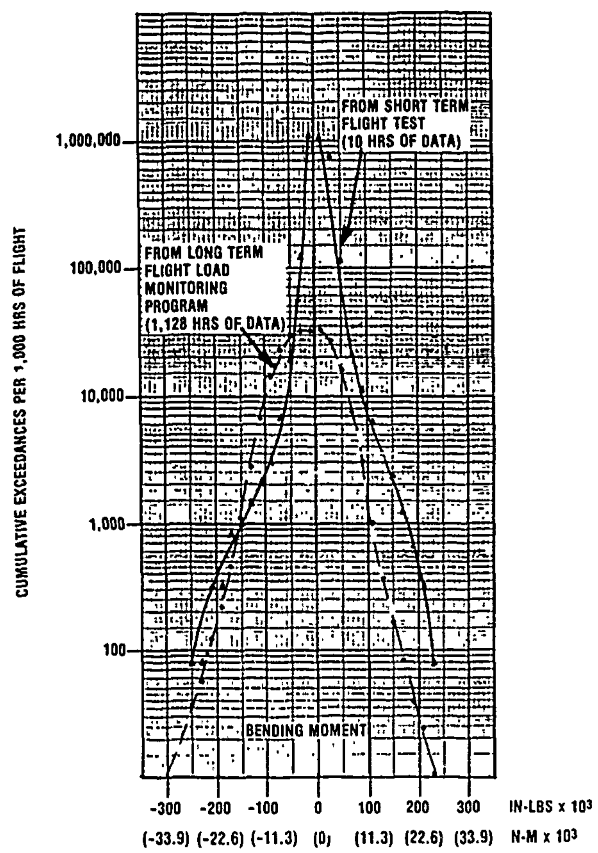


FIGURE 9 — COMPARISON OF BENDING MOMENT SPECTRA OBTAIN FROM SHORT AND LONG TERM FLIGHT LOAD MONITORING PROGRAMS

F-16 FORCE MANAGEMENT, YESTERDAY, TODAY AND TOMORROW

by
R. L. CULP
Engineer
F-16 Loads and Criteria
General Dynamics Corporation
Fort Worth Division
P. O. Box 748 MZ 2802
Fort Worth, Tx USA 76101

SUMMARY

United States Air Force regulations specify that a Force Management Program will be established for each aircraft system within its inventory. This paper presents the approach taken by General Dynamics to fulfill the objectives of the Force Management Program for the F-16 aircraft. The methodology employed, the current status and the future plans for the F-16 Force Management Program in the areas of operational data acquisition, processing and airframe structural maintenance planning are discussed in detail.

1. INTRODUCTION

The overall requirements for aircraft force management programs are given in Air Force Regulation 80-13 which established the Air Force Aircraft Structural Integrity Program (ASIP).

The basic concept of ASIP, as given in AF Reg 80-13, Reference 1, is a "time-phased set of required actions to be performed at the optimum time during the life cycle of an aircraft system, to insure that the aircraft's service life capability is at least equal to its required service life."

The objectives of the ASIP are to:

- Establish, evaluate, and substantiate the structural integrity (airframe strength, rigidity, damage tolerance and durability) of the aircraft.

- Acquire, evaluate, and utilize operational usage data to provide a continual assessment of the in-service integrity of individual aircraft.

- Provide information for decisions regarding force structural planning, modification priorities, and related operational and support decisions.

- Provide a basis to improve structural criteria and methods of design, evaluation, and substantiation for future aircraft systems.

The detailed technical requirements for the ASIP are contained in MIL-STD-1530A, Reference 2, and are made up of five major tasks. The first three tasks are concerned with design information, design analyses and development tests, and full scale testing. The last two tasks, ASIP Tasks IV and V, are defined as the Force Management Program and are concerned with developing force management data packages, collecting operational load indicators and usage data, and operating force management procedures to ensure safe structural service life of individual aircraft during the aircraft's operational phase.

This paper specifically addresses the particular methods of force management selected for the United States Air Force F-16 aircraft, in the past, at present, and in the future. Section 2 provides background information as to the prescribed methodology to accomplish the elements of ASIP Task IV. Section 3 discusses the current utilization of operational data as required for Task IV and the data management transfer effort necessary for implementation of ASIP Task V. Section 4 presents the plans for F-16 force management in the future.

2. FORCE MANAGEMENT METHODOLOGY BACKGROUND

To accomplish the objectives of ASIP Task IV for the F-16 aircraft, the contractor, General Dynamics, is required to provide the Air Force with a force management data package. This package is required to be adequate to manage force operations and maintenance planning during Air Force operation of the force management program during ASIP Task V. Table 2.1 lists the elements of Tasks IV and V as defined by MIL-STD-1530A. The following sections discuss the methodology developed to accomplish each of the elements prescribed in ASIP Task IV for the F-16 Force Management Program (FMP).

TABLE 2.1
USAF AIRCRAFT STRUCTURAL INTEGRITY PROGRAM TASKS
(MIL-STD-1530A)

TASKS TO ASSESS DESIGN AND PLAN STRUCTURAL OPERATION AND MAINTENANCE	
CONTRACTOR TASK IV FORCE MANAGEMENT DATA PACKAGE	AIR FORCE TASK V FORCE MANAGEMENT
FINAL ANALYSIS	LOADS/ENVIRONMENT SPECTRA SURVEY
STRENGTH SUMMARY	INDIVIDUAL AIRPLANE TRACKING DATA
FORCE STRUCTURAL MAINTENANCE PLANS	INDIVIDUAL AIRPLANE MAINTENANCE TIMES
LOADS/ENVIRONMENT SPECTRA SURVEY	STRUCTURAL MAINTENANCE RECORDS
INDIVIDUAL AIRPLANE TRACKING PROGRAM	

2.1 Final Analysis

During the design program for the F-16 aircraft, structural static strength design loads and stress transfer functions were based on accepted analytical methods and evaluation of empirical data from wind tunnel and computer simulated testing. An initial update of the design loads analysis was accomplished upon completion of the F-16 full-scale development (FSD) structural laboratory and flight test programs.

Concurrent with the initial update of the design loads analysis, a flight-by-flight randomized gust and maneuver loads spectrum was developed for use in durability and damage tolerance analysis (DADTA). The updated F-16 Design Loads Spectrum was a combination of two detailed efforts:

1. A Design Loads Library was developed incorporating;

   - wind tunnel steady-state aerodynamic data
   - predictions of static aeroelastic and dynamic loads
   - analog-digital hybrid computer simulated aircraft maneuver response data
   - limited YF-16 prototype flight-measured loads data

2. A Design Usage Model was defined as;

   - 8000 hour service life over 15 years
   - 5776 sorties with 6592 landings (816 touch and go)
   - mission category mix;
     - 55.5% air-to-air missions
     - 20.0% air-to-ground missions
     - 24.5% general missions

Based on the durability and damage tolerance analyses of the F-16 Design Loads Spectra, rational criteria were developed to establish the inspection and repair scheduling and procedures for the F-16 aircraft for the initial operational usage. These criteria considered the types of material and construction, critical crack lengths, inspection capabilities and limits of repair of the aircraft structure.

The objectives of the durability and damage tolerance analyses were to:

- evaluate the durability and stress corrosion cracking of the airframe structure

- evaluate primary safety-of-flight structure for damage tolerance requirements and to identify those in-service inspections required to ensure aircraft safety.

To accomplish these objectives, DADTA methods of damage tolerance crack growth analysis, assuming a typical initial material and/or manufacturing flaw, were used. To assess the effects of usage on the crack growth predictions, a parametric study was conducted by varying the design usage mission mix model. The results of these analyses were then used to prepare the initial Force Structural Maintenance Plan, discussed in Section 2.3, and provided a crack growth library for subsequent use in the Individual Airplane Tracking Program described in Section 2.5.

The Final Analysis element of the FMP requires that after the aircraft have been operational for a specified period of time (three years for the F-16) a Baseline Operational Loads Spectrum will be derived from operational data collected during the Loads/Environment Spectra Survey discussed in Section 2.4. The durability, damage tolerance and service life analysis will be repeated using this baseline spectrum to update inspection and modification requirements for the airframe structural components. A description of the methods employed to accomplish this task is presented in Section 3.

2.2 Strength Summary

The strength summary describes structural design and certification accomplishments for the F-16 airframe and landing gear. It includes structural design criteria, structural arrangement, materials, design load conditions, damage tolerance critical areas, structural margins of safety, and structural test results. The strength summary includes any restrictions for service operations and provides a basis for determining the practicality of operational service different from the design requirements.

2.3 Force Structural Maintenance Plan

Regulations in MIL-STD-1530A under Task IV require that the contractor "prepare a Force Structural Maintenance Plan (FSMP) to identify the inspection and modification requirements and the estimated economic life of the airframe."

The initial FSMP for the F-16 airframe was based on the results of all analyses and structural tests conducted during the F-16 Full Scale Development Program. The essential elements of the initial FSMP are:

- Maintenance Activity Summary

- Detailed Structural Inspection and Maintenance Requirements

- Analysis and Test History Summary

- Potential Structural Maintenance Areas

- Adjustment Procedures based on Individual Airplane Tracking Results

The FSMP provides guidelines for conducting specific maintenance actions. The plan specifies applicable aircraft, inspection intervals, repair guidelines, and cost data in terms of man hours and downtime. The Air Force uses this plan for budgetary, force structure, and maintenance scheduling and planning.

2.4 Loads/Environment Spectra Survey

During the final analysis stage of Task IV, a design usage model, including flight and maneuver profiles, was selected for the aircraft. Since the actual usage of the aircraft may impose a load and stress environment on the aircraft that is different from the design, the FMP requires an early assessment of the effects of operational usage at critical locations. This element of the FMP is obtained through the Loads/Environment Spectra Survey (L/ESS).

MIL-STD-1530A defines the objective of the L/ESS is "to obtain time history records of those parameters necessary to define the actual stress spectra for the critical areas of the airframe." It is the responsibility of the contractor to determine the required parameters to be obtained, the instrumentation requirements, the number of aircraft to be instrumented, and the length of the recording period. The contractor must design a data processing system for collection and processing of these time history records. The system must be compatible with current Air Force capabililities of the Aircraft Structural Integrity Management Information System (ASIMIS) since one of the requirements of Task V is a transfer of FMP data processing procedures to ASIMIS.

2.4.1 L/ESS Data Acquisition

To support the data acquisition portion of L/ESS for the F-16 FMP, the MXU-553/A airborne Flights Loads Recorder (FLR) system was selected for use on every sixth aircraft. The FLR system consists of a signal data recorder, a signal data converter/multiplexer, and associated sensors. The recorder and converter/multiplexer units are located in the left side aft equipment bay of the aircraft to facilitate easy inspection of the units and replacement of the recorded data cartridge.

The FLR unit receives, processes and records signals from control surface position and engine power lever angle transducers, the flight control computer, central air data computer, central interface unit of the stores management subsystem, engine core (N2) RPM indicator, fuel quantity indicator, and an electrical strain gage located in the aft fuselage of the aircraft. Certain "documentary data" values such as aircraft serial number, date, mission type, stores description codes, and weights are input via thumb wheels in the front of the recorder unit before each flight.

The recorder is automatically turned on when the throttle is first advanced following engine start and runs continuously until the engine is shut down. This continuous operation provides time history records of the recorded parameters for both ground and flight operations. Table 2.2 lists the recorded parameters, sampling rate and source along with additional parameters calculated from the FLR data.

The FLR recording medium is a replaceable tape cartridge with a capacity of 15 hours. To insure that a majority of in-flight data is recorded, the FLR cartridge is replaced at 10-hour intervals by field maintenance personnel, then transmitted to ASIMIS for processing as described in the following section.

TABLE 2.2
F-16 LOADS/ENVIRONMENT SPECTRA SURVEY PARAMETERS

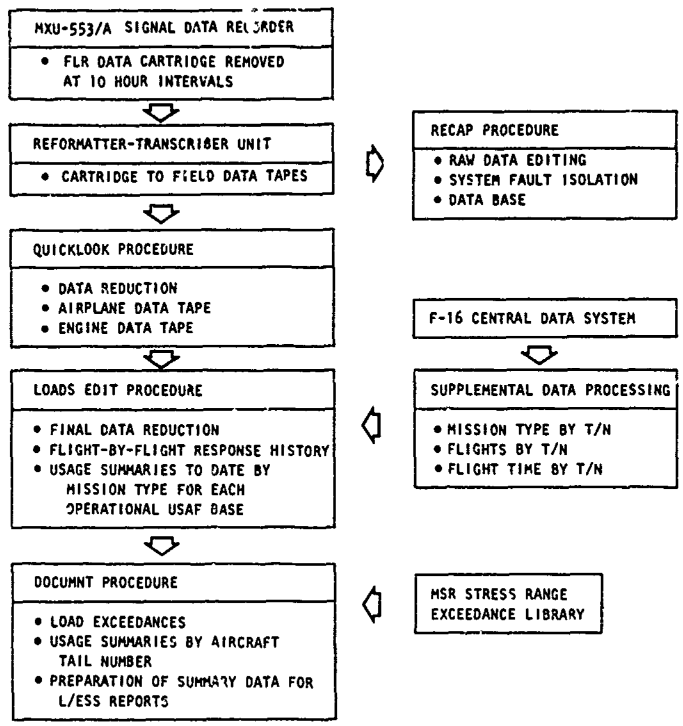
PARAMETER (SYMBOL/ACRONYM)		SAMPLING RATE/SEC	SOURCE	PROGRAM CALCULATED PARAMETERS	
PRESSURE ALTITUDE	(Hp)	1	CADC	TIME-IN-FLIGHT	QL
CALIBRATION AIRSPEED	(Vc)	1	CADC	GROSS WEIGHT	QL/LE
PITCH RATE	(Q)	15	FCC	MACH NUMBER	QL
YAW RATE	(R)	15	FCC	DYNAMIC PRESSURE	QL
ROLL RATE	(P)	30	FCC	RIGHT HT BENDING MOMENT	LE
ROLL ACCELERATION	(Pdot)	30	FLR C/M	VERTICAL TAIL BENDING	LE
VERTICAL ACCELERATION	(Nz)	15	FCC	LEFT WING ROOT BENDING	LE
LATERAL ACCELERATION	(Ny)	15	FCC	RIGHT WING ROOT BENDING	LE
LONGITUDINAL ACCELERATION	(Nx)	5	ACCELEROMETER	LEFT HT BENDING MOMENT	LE
FUEL QUANTITY	(FQ)	1	INDICATOR	PEAK INDICATOR CODE	QL/LE
ENGINE ROTOR SPEED	(N2)	1	INDICATOR	CENTER OF GRAVITY	LE
RUDDER POSITION	(DR)	15	LVDT	STORES CONFIGURATION	LE Doc
LEFT HT POSITION	(DHL)	15	LVDT		
RIGHT HT POSITION	(DHR)	15	LVDT		
LEFT FLAPERON	(DFL)	15	LVDT		
RIGHT FLAPERON POSITION	(DFR)	15	LVDT		
POWER LEVER ANGLE	(PLA)	5	LVDT		
STRUCTURAL STRAIN		15	STRAIN GAGE		
EVENTS SIGNAL		1	FLR		
LG DOWN CMD					
WEIGHT-ON-WHEELS					
WEAPON RELEASE					

CADC = Central Air Data Computer
FCC = Flight Control Computer
FLR C/M = Flight Loads Recorder Converter/Multiplexer
LVDT = Linearily Variable Differential Transducer
QL = Quicklook Computer Procedure
LE = Loads Edit Computer Procedure
HT = Horizontal Tail

2.4.2 L/ESS Data Processing

Four major steps are used in the processing of the FLR data for the L/ESS function of the F-16 FMP. One procedure is a standard procedure developed by ASIMIS for use on all MXU-553/A flight loads recorder systems, while the other three were developed by General Dynamics for the F-16 FMP. Figure 2.1 illustrates the flow of data processing from the removal of the FLR data cartridge to the outputs required for the L/ESS.

FIGURE 2.1
F-16 L/ESS DATA PROCESSING



After removal from the operational aircraft, the FLR data cartridge is sent to ASIMIS for processing. The cartridge is first run through a reformatter/transcriber (R/T) unit where the recorded data is transcribed onto a magnetic computer tape. The R/T tapes are further processed through a computer procedure called RECAP, which was developed by ASIMIS for the evaluation of all MXU-553/A signal data recorder R/T data tapes. Analysis includes reviewing the data to identify recorder, multiplexer, or sensor malfunctions and to verify that the data is acceptable for subsequent data processing. Output of RECAP includes flight headers, summaries, events, histograms, and fault isolation lists.

The initial processing of the R/T tape is accomplished through the Quicklook (QL) procedure. This procedure is very similar to RECAP in that output consists of flight headers, summaries, events, histograms, and a list of possible data spikes. In addition, a table of percent-of-time by Mach number, altitude, and gross weight is calculated for each flight. On-ground and in-flight time periods are distinguished and periods of significant in-flight maneuver activity are identified for use in the L/ESS and subsequent data processing. The QL procedure also calculates four additional parameters from the FLR data samples: time in flight, gross weight, Mach number, and dynamic pressure. A time history tape of the FLR data, modified to exclude certain non-maneuvering time periods and to add the four calculated parameters, is then output for further processing.

The QL procedure also produces a tape for use in the development of engine usage spectra. This tape, termed the PLA tape, contains the time histories of four FLR parameters, Mach, Altitude, Engine Rotor Speed and Engine Power Lever Angle, at one sample per second and a fifth parameter, vertical acceleration (Nz) at fifteen samples per second. The PLA tape is used to evaluate the operational engine usage in a manner similar to the structural analyses.

Before processing of the FLR data, the QL flight header records are validated by showing agreement with aircraft tail number, mission type, and stores configuration information obtained through the F-16 Centralized Data System of the USAF corresponding to the specific flight period of the FLR data. Valid QL tapes are then processed through the Loads Edit procedure.

Procedure Loads Edit performs final reduction of flight-by-flight FLR data and accumulates tables of control surface position change occurrences, exceedances of vertical acceleration (Nz), lateral acceleration (Ny), roll rate (P), and roll acceleration (Pdot), and landing gear events. These tables, along with the percent-of-time tables read in from the QL output tape are referred to as summary tables. Procedure Loads Edit maintains a data base containing cumulative summary tables to date for each operational USAF base from which data is regularly received.

Loads Edit uses equations derived through regression analysis of structural loads flight test data to compute values of left and right horizontal tail and vertical tail bending moment throughout each maneuver. The FLR data samples are reduced using a peak-valley data compression routine based on the activity of these three load parameters, the response parameters P and Pdot, and the vertical acceleration Nz. The reduced FLR data is output onto a magnetic tape, called the compressed time history (CTH) tape, along with the summary tables for each flight. The cumulative summary tables are output onto a separate magnetic tape.

Procedure DOCUMNT processes the CTH and cumulative summary table tapes output by Loads Edit to produce exceedance tables of Nz separated by aircraft tail number, mission type, and gross weight range. These tables are subsequently published in the periodic L/ESS reports as required for the FMP. DOCUMNT computes stress exceedances based on FLR data for comparison with Mechanical Strain Recorder stress exceedance data compiled by the Individual Airplane Tracking analysis procedure further discussed in Section 2.5. In addition, DOCUMNT accumulates percent-of-time tables for all mission types for which data has been received to date.

Procedure DOCUMNT is the final step of the L/ESS data analysis. The CTH tapes produced during the L/ESS are accumulated and later used for development of operational loads spectra.

2.4.3 Baseline Operational Loads Spectrum

Upon completion of the collection and processing of three years of FLR data, the contractor is required to develop a Baseline Operational Loads Spectrum for updating durability and damage tolerance analyses and subsequent structural maintenance planning actions. This loads spectrum incorporates the variation in the usage of the aircraft encountered during the three year period of the L/ESS. The current status of the F-16 FMP includes analyses of the Baseline Operational Loads Spectrum. A description of the development and processing of the FLR data to accomplish this task is given in Section 3.

2.5 Individual Airplane Tracking Program

It is stated in MIL-STD-1530A that the objective of the Individual Aircraft Tracking (IAT) program shall be "to predict the potential flaw growth in critical areas of each airframe that is keyed to damage growth limits of MIL-A-83444, inspection times, and economic repair times." To accomplish this portion of the FMP, appropriate usage parameters, data collection methods, and data analysis programs are developed by the contractor for later transfer (during implementation of Task V) to ASIMIS. The following sections discuss the particular equipment and methods employed by General Dynamics to accomplish this task for the F-16 FMP. Figure 2.2 illustrates the data collection and processing procedures.

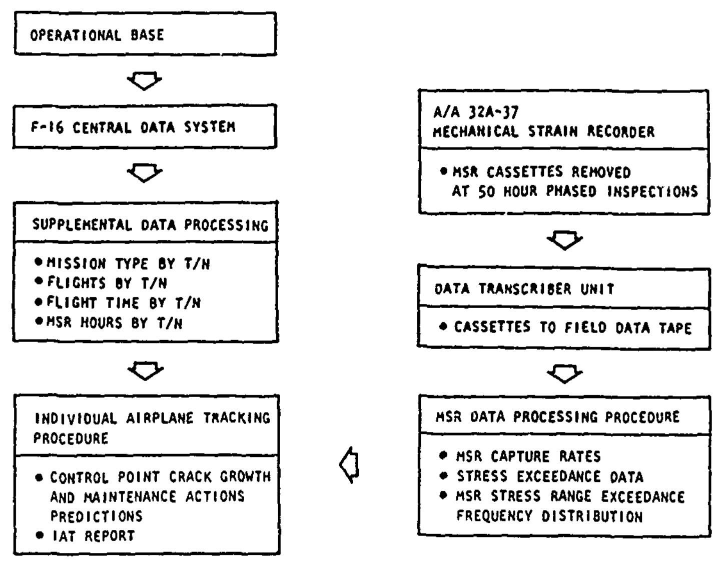
2.5.1 IAT Data Acquisition

To monitor and record the required tracking data for each F-16 aircraft, General Dynamics chose to install the A/A 32A-37 Mechanical Strain Recorder (MSR). The MSR system is self-contained, requiring no electrical power, and consists of the recorder assembly, a data cassette, and a protective cover. The MSR is installed in each aircraft on the lower right-hand flange of a center fuselage wing carry-through bulkhead. The system is easily accessible through the main landing gear wheel well. The strain at this location is predominately sensitive to wing root bending moment, and is used for defining the individual aircraft's maneuver activity time history.

The MSR has two diamond styluses which inscribe traces onto the metallic recording tape of the data cassette. One trace is used as a reference line while the other trace defines the strain levels caused by aircraft maneuvers.

The MSR cassette's recording capacity is a function of the strain activity of the fuselage bulkhead since the tape is advanced proportionally to the severity of the strain cycle occurring during maneuvers. The capacity is between 70 and 100 flight hours for the operational aircraft. To allow for variations in maneuver activity, the replacement interval is set at 50 hours. At the 50 hour interval the cassette is replaced and transmitted to ASIMIS for transcription to a digital computer data tape for subsequent processing in the IAT program.

FIGURE 2.2
F-16 IAT DATA PROCESSING



2.5.2 IAT Data Analysis

The F-16 IAT data analysis methods utilize the concept of usage variations encompassing the range of aircraft usage from the least severe to the most severe. The individual aircraft unique usage is compared with these usage variations to predict crack growth rates at selected tracking points of the airframe structure. The required usage variation crack growth curves are derived from durability and damage tolerance analyses of loads spectra developed during a study conducted specifically for this purpose. An update of these crack growth curves, based on operational usage collected during the L/ESS and field inspections, is required for each of the selected tracking points, listed in Table 2.3.

The initial data processing procedure for the MSR data tape converts the sequential raw strain data to valley-to-peak stress histories at the MSR location. These stress histories are then processed, as outlined in the following steps, to define the individual aircraft's unique usage during the specified time period.

1. From the stress histories recorded by the MSR for a specific aircraft, identified by tail number (T/N), a stress cyclic-range squared ($\Delta\sigma^2_{MSR}$) exceedance distribution per 1000 hours is developed.

2. Slope and intercept (m and b) values are derived for the exceedance distribution using an exponential (exp) curve fit routine to arrive at an equation of the form:

$$N(\Delta\sigma^2_{MSR})_{T/N} = \exp\left(m_{T/N}\Delta\sigma^2_{MSR} + b_{T/N}\right)$$

   where N = Number of exceedances

3. The m and b values are stored on a magnetic tape with corresponding aircraft tail number and reporting period for use in the IAT analysis procedure.

Before processing of the MSR derived exceedance data for a specified reporting period, supplementary data of aircraft usage at each base, as collected by the F-16 Centralized Data System (CDS), must be reviewed and processed. This usage data, collected on a daily basis and transmitted monthly to General Dynamics, consist of:

- base of operation
- airplane tail number
- date of flight
- number of flight hours
- mission type for the flight

The CDS data provides information to the IAT analysis procedure to account for total number of flights, total flight hours and total flight hours by mission type accumulated for each aircraft. This data may also be used to account for inadequate or missing MSR data for each aircraft during the specified reporting period. This supplementary data is further used to complete the usage summary tables of the IAT periodic reports required for the F-16 FMP.

The IAT analysis procedure uses the MSR stress exceedance data, the usage data provided by CDS, and a flaw growth library derived from the five variations of the initial F-16 Design Loads Spectrum to create cumulative flaw growth curves for each IAT control point on each individual aircraft. The flaw growth library contains predicted crack growth curves for each of the IAT control points and normalized $\Delta\sigma^2_{MSR}$ equations for each of the five usage variations. The methods used to calculate and normalize the $\Delta\sigma^2_{MSR}$ exceedance distribution equations for the five usage variations are shown in Figure 2.3. Figure 2.4 outlines the steps utilized in the IAT analysis procedure to predict crack growth at each control point for each aircraft during a specified reporting period.

The IAT analysis procedure outputs a magnetic tape containing the accumulated flaw growth values for the individual aircraft and a printout of the flaw growth, the maintenance action times, and the aircraft usage summaries which go into periodic IAT reports. The IAT reports, published at six month intervals, provide the data necessary to modify the inspection and repair schedules for the critical areas of the airframe structure on each individual aircraft.

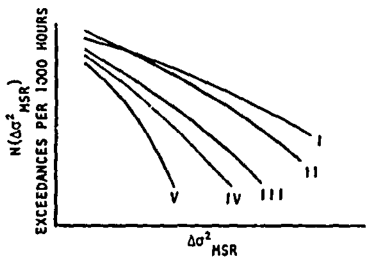
TABLE 2.3
F-16 INDIVIDUAL AIRPLANE TRACKING CONTROL POINTS

<u>WING STRUCTURE</u>
UPPER WING SKIN CUTOUT AT BL 61.5
WING ROOT LOWER ATTACH FITTING
LOWER WING SKIN PYLON CUTOUT AT BL 71
LOWER WING SKIN FUEL VENT AT BL 102
FRONT SPAR LOWER FLANGE AT LEF HINGE #2
LOWER WING SKIN FASTENER HOLES AT BL 120
LOWER WING SKIN SURFACE AT BL 112, HIGH YIELD MATERIAL
<u>FORWARD FUSELAGE STRUCTURE</u>
COCKPIT SILL LONGERON
<u>CENTER FUSELAGE STRUCTURE</u>
WING SUPPORT BULKHEAD SHEAR WEB
BULKHEAD AT FS 341.8, REFUELING WELL STIFFENER FASTENER HOLES
UPPER BULKHEAD WEB FILLET RADIUS AT WING ATTACH
<u>AFT FUSELAGE STRUCTURE</u>
BULKHEAD AT FS 479.8, UPPER FLANGE BOLT HOLE AT BL 23
<u>VERTICAL TAIL STRUCTURE</u>
VERTICAL TAIL ATTACH FITTING WEB-PAD RADIUS

FIGURE 2.3

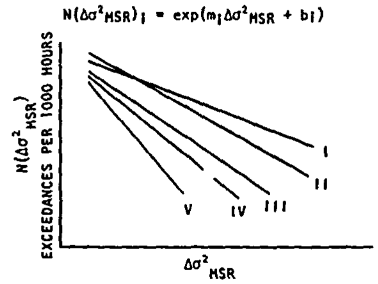
USAGE VARIATION CRACK GROWTH PREDICTIONS FOR IAT PROGRAM

STEP 1: GIVEN STRESS TIME HISTORIES AT THE MSR LOCATION BASED ON DESIGN LOADS SPECTRA, DERIVE $\Delta\sigma^2_{MSR}$ EXCEEDANCE DISTRIBUTIONS FOR EACH OF THE 5 USAGE VARIATIONS.

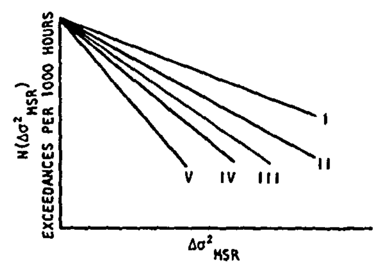


STEP 2: COMPUTE AN EXPONENTIAL CURVE FIT OF EACH OF THE $\Delta\sigma^2_{MSR}$ EXCEEDANCE DISTRIBUTIONS BY LINEAR REGRESSION TO DERIVE AN EQUATION, WHERE I = I, II, III, IV, V, IN THE FORM:

$$N(\Delta\sigma^2_{MSR})_i = \exp(m_i \Delta\sigma^2_{MSR} + b_i)$$



STEP 3: NORMALIZE $\Delta\sigma^2_{MSR}$ EXCEEDANCE DISTRIBUTIONS TO THE LARGEST "b" VALUE FOUND IN STEP 2 BY MULTIPLYING $N(\Delta\sigma^2_{MSR})_i$ BY $\exp(b_{II}-b_i)$ (ASSUMING b_{II} IS LARGEST)



STEP 4: DETERMINE NORMALIZED TIME CRACK GROWTH CURVES FOR EACH TRACKING CONTROL POINT FOR EACH USAGE VARIATION BY MULTIPLYING CRACK GROWTH TIME BY $\exp(b_i - b_{II})$.

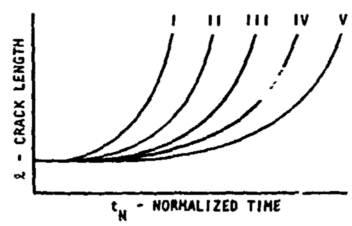
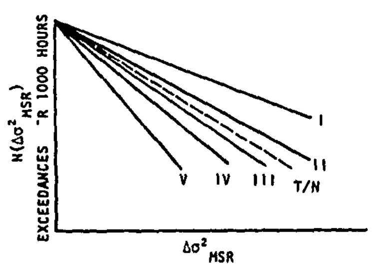


FIGURE 2.4

INDIVIDUAL AIRPLANE TRACKING PROGRAM DATA ANALYSIS

STEP 1: GIVEN A NORMALIZED $(\Delta\sigma^2_{MSR})T/N$ EQUATION FOR AN AIRCRAFT FOR A REPORTING PERIOD, DETERMINE AN INTERPOLATION CONSTANT BY:

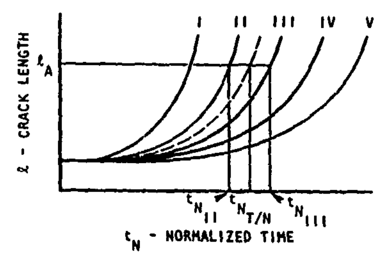
$$IC = \frac{m_{II} - m_{T/N}}{m_{II} - m_{III}} \quad \text{ASSUMING } m_{III} < m_{T/N} < m_{II}$$



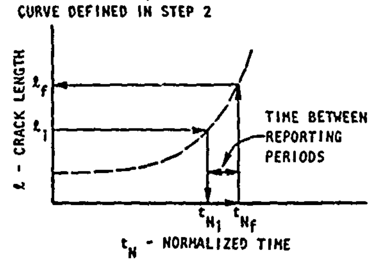
STEP 2: DETERMINE THE NORMALIZED TIME CRACK GROWTH CURVE FOR EACH CONTROL POINT FOR THE AIRCRAFT BY VARYING CRACK LENGTH (ℓ):

AT A GIVEN ℓ (ℓ_A)

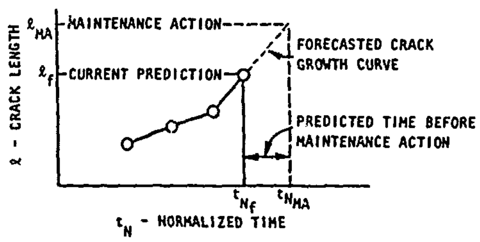
$$t_{N_{T/N}} = t_{N_{II}} + IC(t_{N_{III}} - t_{N_{II}})$$



STEP 3: DETERMINE INCREMENTAL CRACK LENGTH (ℓ_f) FOR EACH CONTROL POINT FOR THE AIRCRAFT FOR THE REPORTING PERIOD GIVEN AN INITIAL CRACK LENGTH (ℓ_i) AND THE CRACK GROWTH CURVE DEFINED IN STEP 2



STEP 4: DETERMINE ACCUMULATED CRACK LENGTH AND MAINTENANCE ACTION FORECAST FOR EACH CONTROL POINT FOR THE AIRCRAFT FROM ALL REPORTING PERIODS.



3. CURRENT FORCE MANAGEMENT STATUS

The requirements in the ASIP for Task V specify that the Air Force will be pri. .rily responsible for utilizing the Force Management Data Package supplied by the contractor during Task IV with a minimum of assistance. To meet this requirement, an overall transfer plan was established corresponding to specific guidelines. The current status of this transfer effort in the areas of L/ESS and IAT are discussed in the appropriate sections following.

Specifications within the Final Analysis and IAT elements of the F-16 FMP require that a Baseline Operational Loads Spectrum, along with variations of the spectrum, be developed to provide updates to durability and damage tolerance analyses and subsequently to the Force Structural Maintenance Plan. Development of these load spectra are now complete or nearing completion. Section 3.3 discusses the methods utilized to accomplish this task.

3.1 Loads/Environment Spectra Survey

The L/ESS for the F-16 aircraft is currently providing the specified time history records and usage information to General Dynamics and the Air Force in a timely and organized manner. The L/ESS have proven that operational data may be collected, processed and utilized to provide a data base for use in defining structural maintenance actions.

The three year operation of the F-16 L/ESS by General Dynamics has produced seven periodic reports documenting the collection of the FLR data and the required usage information. The L/ESS data processing procedure Quicklook, described in Section 2.4.2. is currently operational at ASIMIS. The Loads Edit procedure is scheduled to be transferred to ASIMIS during 1984.

As of January 1, 1984 a total of approximately 2100 flights containing 2500 hours of FLR data have been accumulated for use in the development of operational loads spectra. These loads spectra include the Baseline Operational Loads Spectrum, required for update of the Design Loads Spectrum developed in the Final Analysis element, and the usage variation loads spectra required for the update of the crack growth models used in the IAT. In addition, the FLR data is proving valuable in the development of loads spectra for variations of the F-16 airframe.

3.2 Individual Airplane Tracking Program

The F-16 IAT program provides predictions for flaw growth and structural maintenance actions at selected control points for each aircraft in the USAF. The accumulated flaw growth predictions, based on individual aircraft MSR defined stress histories or the individual aircraft usage data provided by CDS for periods of incomplete or missing MSR data, call for initial maintenance actions to occur in the 1990 time frame on the average.

The F-16 IAT data processing procedures were operated by General Dynamics through 1982, producing six periodic reports. In 1983 the procedures were transferred to ASIMIS for Air Force operation.

3.3 Baseline Operational Loads Spectrum

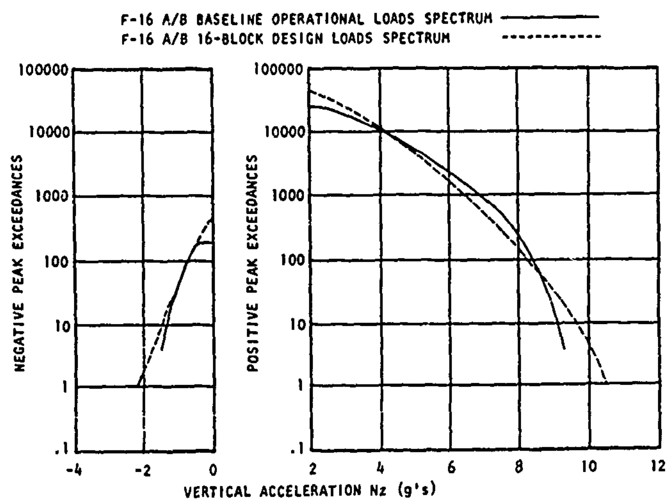
The development of operational loads spectra from the FLR data base begins with the selection of appropriate FLR flights to recreate the operational usage encountered by the aircraft in the field. The accumulated operational usage mission mix at three primary USAF F-16 bases and the flight time weighted composite mission mix computed for the baseline operational loads spectrum are shown in Table 3.1. The F-16 design usage mission mix is also given in Table 3.1 for comparison.

TABLE 3.1
F-16 L/ESS USAGE MISSION MIX

MISSION CATEGORY	USAF BASE 1	USAF BASE 2	USAF BASE 3	COMPOSITE USAGE	DESIGN USAGE
AIR-TO-AIR	35.3%	35.6%	26.3%	32.8%	55.5%
AIR-TO-GROUND	39.9%	30.2%	60.2%	42.4%	20.0%
GENERAL	24.8%	34.2%	13.5%	24.8%	24.5%

The selection of the FLR flights for the operational loads spectrum must also satisfy the requirement that the maneuver activity of the spectrum meet the composite vertical acceleration (Nz) exceedance distribution of the L/ESS. Figure 3.1 illustrates the comparison of the Nz exceedance distribution provided by L/ESS Report No. 6 for development of the F-16 Baseline Operational Loads Spectrum with that of the F-16 Design Loads Spectrum.

FIGURE 3.1
F-16 PEAK LOAD FACTOR EXCEEDANCE DISTRIBUTION PER 1000 HOURS



The concept of a repeatable block of flight-by-flight loads histories is utilized in the development of the operational loads spectrum. A 500 hour block of FLR data is selected representing the baseline usage and repeated to build the 8000 hour service life loads spectrum. This method is beneficial in that it reduces some analysis expense and it enhances fractography results from laboratory test specimens.

After selection of the FLR flights for use in the operational loads spectra, the edited FLR data for each flight are processed to compute load time histories at selected control points. The methods employed in the prediction of loads from the FLR data have been updated from the methods developed in early FMP analyses. Current FMP load equations are derived from various sources, including:

- F-16 FSD and follow-on structural flight test data
- analytical design loads
- updated wind tunnel data

The development of load equations from the structural flight test data utilize linear regression analysis techniques to arrive at equations of the form:

$$\text{Load} = C_0 + C_1 * (P_1) + C_2 * (P_2) + \ldots + C_n * (P_n)$$

where $C_1 - C_n$ are the regression coefficients

and $P_1 - P_n$ are appropriately selected FLR parameters

Development of loads predictions from analytical design loads and wind tunnel aerodynamic load coefficient data normally is accomplished using interpolation methods on a finite grid of predicted loads. Due to higher cost, in terms of computer processing time, this method is only used when the required load items are not available from the flight test data or the loads equations derived from regression analysis prove inadequate. In certain cases, the analytical or wind tunnel derived loads predictions grid is sufficient to process through regression analysis to provide loads equations with a very high degree of correlation, thus, reducing computer processing costs.

Upon completion of the processing of all selected FLR flights in the 500 hour block, documentation, consisting of maximum/minimum loads, total flight time, percent-of-time by Mach number and altitude, mission type summaries for usage definition, and exceedance distributions of significant maneuver response and load parameters are output. The computed loads time histories provide the basis for the durability and damage tolerance analyses at the selected structural control points.

At this time, the F-16 A/B Baseline Operational Loads Spectrum is being evaluated by durability and damage tolerance analyses procedures. Laboratory testing of items in the structural evaluation program update is progressing. The usage variation spectra, to support the IAT program, are currently being developed for completion in 1984.

3.4 F-16 C/D Loads Spectra

The FLR data, accumulated through the L/ESS of the F-16 A/B operational aircraft, was first utilized in the development of a F-16 C/D Advance Loads Spectrum. The development of this loads spectrum proved that the FLR data could be used for spectra development for F-16 model variations. However, there were problems to overcome when predicting loads on a variation with certain system configuration changes. It was found that operational differences from the baseline F-16 A/B systems, such as the leading edge variable camber system, produce model peculiar loads on the airframe. To alleviate these problems, methods were developed incorporating stability and control and structural flight test data and additional analytical loads predictions.

The methods for processing FLR data for development of operational loads spectra, described in Section 3.3, and the methods mentioned above are currently being utilized to develop an initial baseline loads spectrum and usage variation spectra for the F-16 C/D aircraft. Updates to these spectra will be provided in the future using the expanded capabilities of the F-16 C/D Force Management Program described in the following section.

4. FUTURE FORCE MANAGEMENT PLANS

Development and integration plans are now being made at the USAF F-16 SPO and General Dynamics for addition of a Crash Survivable Flight Data Recorder (CSFDR) System to be incorporated into the advanced avionics system of each F-16 C/D aircraft. The specification for this new system contains Tri-Service (Air Force, Army, and Navy) requirements for a common flight data recorder. The F-16 C/D CSFDR system consists of three hardware units:

- Signal Acquisition Unit (SAU)
- Crash Survivable Memory Unit (CSMU)
- Auxiliary Memory Unit (AMU)

The SAU, to be located in the aft left equipment bay, will receive signals from the aircraft multiplex bus and from other data sources. Then, through the use of a microcomputer, the SAU will process, convert and compress the analog signals to digital output on a realtime basis. This output will be saved within the SAU, stored in the CSMU, or stored in the AMU which will be mounted within the SAU. The SAU and AMU are not required to survive severe crashes. The CSMU will be located in the aft fuselage area of the aircraft for improved survivability.

Four types of data will be processed by the SAU:

1. Mishap Investigation data
2. Individual Airplane Tracking data
3. Structural Loads Environment data
4. Engine Usage data

Although the actual on-board data processing algorithms and data storage medium for each data type may differ, there is a commonality of input parameters to be processed by the SAU. Table 4.1 gives the input/output parameters of the SAU. Also included are the sampling rate, data type use, and notes as to where the parameter signal will be obtained. As can be seen, most signals will be readily available from the Flight Control Computer, or the aircraft systems multiplex bus. With the exception of the control surface position transducers, all CSFDR sensors will be included in the basic F-16 C/D system.

The CSMU wil˜ receive data from the SAU and will maintain a minimum of 15 minutes of aircraft flight data in a nonvolatile memory medium. The CSMU will have very stringent operational and survivability requirements.

The four types of data to be processed and stored by the CSFDR system are addressed in the following sections. The on-board processing methods, memory capabilities, and anticipated post-flight processing and utilization methods for each data type are discussed.

TABLE 4.1

F-16 CRASH SURVIVABLE FLIGHT DATA RECORDER SYSTEM INPUT/OUTPUT PARAMETERS

PARAMETER (SYMBOL/ACRONYM)	SAMPLING RATE/SEC	DATA TYPE USE	SOURCE
----- FLIGHT CONDITIONS			
CORRECTED ALTITUDE (HP)	1	1 2 3 4	MUX
CAL AIRSPEED (VC)	1	1 3	MUX
TRUE FREESTREAM AIR TEMP	1	1 3 4	MUX
ANGLE OF ATTACK	8	1 3	MUX
RADAR ALTITUDE	1	1	MUX
RADAR ALT LOW SET	1	1	MUX
TRUE HEADING	4	1	MUX
HSI COURSE DEV	1	1	MUX
PITCH ATTITUDE	4	1	MUX
ROLL ATTITUDE	4	1	MUX
GROSS WEIGHT (GW or W)	1	2 3	MUX
MACH NUMBER (M)		3 4	SAU Calc
----- AIRCRAFT RESPONSE			
PITCH RATE (Q)	16	2 3	FCC Analog
PITCH ACCEL (Qdot)	16	3	SAU Calculation
ROLL RATE (P)	16	2 3	FCC Analog
ROLL ACCEL (Pdot)	16	3	SAU Calculation
YAW RATE (R)	16	2 3	FCC Analog
YAW ACCEL (Rdot)	16	3	SAU Calculation
LONG ACCEL (Nx)	4	1 3 '	FCC Analog
VERT ACCEL (Nz)	16	1 2 3 4	FCC Analog
LAT ACCEL (Ny)	8	1 3	FCC Analog
----- CONTROL SURFACE POSITIONS			
RUDDER POSITION (DR)	8	1 3	LVDT
LEFT HT POS (DHL)	16	1 3	LVDT
RIGHT HT POS (DHR)	16	1 3	LVDT
LEFT FLAPERON POS (DFL)	16	1 3	LVDT
RIGHT FLAPERON POS (DFR)	16	1 3	LVDT
LEF POSITION (DLEF)	4	1	LEF Feedback
----- PILOT INPUTS			
STICK FORCE - LONG	4	1	FCC
STICK FORCE - LATERAL	4	1	FCC
----- ENGINE PARAMETERS			
POWER LEVER ANGLE (PLA)	2	1 4	Hard wired
CORE RPM (N2)	2	1 4	Indicator
FAN RPM (N1)	2	1 4	N1 Signal
FAN TURBINE INLET TEMP	2	1 4	Indicator
NOZZLE POSITION	1	1	Sync Indicator
TOTAL FUEL QUANTITY (FQ)	1	1 2 3	MUX
FWD FUEL QUANTITY (FFQ)	1/5	1	FLC
AFT FUEL QUANTITY (AFQ)	1/5	1	FLC
FUEL FLOW (FF)	1	1	FF Meter

PARAMETER	SAMPLING RATE/SEC	DATA TYPES
-----DISCRETE SIGNALS		
ELEC ENG CONTROL	1	1
JFS "START 2"	1	1
AIR LIGHT (EPU)	1	1
GEN - MAIN FAIL	1	1
GEN - STBY FAIL	1	1
FLCS PWR TEST SWITCH	1	1
FWD FUEL RES LOW	1	1
AFT FUEL RES LOW	1	1
HYD PRESS "A" LOW	1	1
HYD PRESS "B" LOW	1	1
FSC CAUTION "RESET"	1	1
AUTO PILOT-ROLL/HDG	1	1
AUTO PILOT-ROLL/ATT	1	1
AUTO PILOT-PITCH/HDG	1	1
AUTO PILOT-PITCH/ATT	1	1
FWD PADDLE DEPRESS	1	1
AFT PADDLE DEPRESS	1	1
CONTROL STICK POS	1	1
CAT I/III POS	1	1 2 3
DUAL FC FAILURE	1	1
AUTO TF SELECT	1	1 3
MANUAL TF SELECT	1	1 3
TF FAILURE LT	1	1
OBS WARN LT	1	1
AUTO TF FAILURE LT	1	1
VALID WEAPON RELEASE	1	1 2 3
SIMULATED WEAPON REL	1	1
MLG WEIGHT-ON-WHEELS	1	1 2 3
LG DOWN COMMAND	2	1 2 3
COMM XMIT (UHF/VHF)	1	1
OVERHEAT	1	1

MUX = Multiplex Bus
SAU = Signal Acquisition Unit
FCC = Flight Control Computer
LVDT= Linearlly Variable Differential Transducer
HT = Horizontal Tail
LEF = Leading Edge Flap
FLC = Fuel Level Control

4.1 Mishap Investigation Data (Type 1)

The retention of Type 1 data in the CSMU will increase the capabilities of mishap investigation for the F-16 C/D aircraft. The evaluation of these data will provide information as to the cause of the mishap, including the flight and aircraft systems conditions.

The SAU will continuously process Type 1 data during flight and record data in the CSMU using algorithms for storing various types of data consisting of:

- Special Events
- Baseline Data
- Continuous Update Data

The special event data will consist of signals recorded 15 seconds before and after the indicated event. Up to five special events may be recorded in any flight. Sixteen different criteria will be programmed into the SAU to trigger the special event algorithm such as:

- pilot command
- signals exceeding limits
- discrete signal indicating a malfunction

The baseline data, consisting of recorded signals 15 seconds before and after lift-off, and the special events data will be stored with overwrite protection in the memory of the CSMU.

The continuous update data will be stored in the CSMU using a logic-state-change criteria for discrete signals and a floating aperture data compression technique for the digital/analog parameter signals. The CSMU will contain sufficient memory capacity to record a minimum of 15 minutes of flight data before the oldest data samples are overwritten. The Type 1 data will be extracted from the CSMU via the SAU as required to support mishap investigation by qualified base personnel using a Memory Loader/Verifier (MLV) unit or directly from the CSMU.

4.2 Individual Airplane Tracking Data (Type 2)

Type 2 data, for the F-16 C/D Individual Airplane Tracking (IAT) program, will be processed and accumulated by the SAU. The accumulation of these data will require a minimum of memory within the SAU since the data will be saved in the form of exceedance and/or occurrence tables rather than a time history required for the other data types.

The inflight processing of the Type 2 data will include the accumulation of exceedance/occurrence tables of vertical acceleration (Nz) and valley-to-peak delta NzW at the aircraft center of gravity. In addition, a cumulative summary of number of touch-downs, number of gear extension events, number of flights, total flight time, and vertical acceleration overloads with associated gross weight will be stored. It is planned that the accumulated data will be collected (but not cleared from memory) at 100-hour phased inspection intervals at the base level using the MLV unit.

Post-flight processing of the Type 2 data will utilize the delta NzW exceedance/occurrence tables in much the same manner as the MSR data of the F-16 A/B IAT program. The addition of the other Type 2 data will provide a more detailed capability for the tracking of the usage of individual aircraft.

The data to be processed and accumulated by the SAU for tracking of individual aircraft are selected to be independent of flight crew inputs and will require only minimal ground personnel interaction. It is anticipated that this will result in high capture rates of individual aircraft usage data and therefore provide increased data sampling for the IAT program for the F-16 C/D aircraft.

4.3 Structural Loads Environment Data (Type 3)

The processing by the SAU and storage in the AMU of the Type 3 data will provide greatly expanded and simplified methods for the Loads/Environment Spectra Survey of the F-16 C/D aircraft. In-flight processing will include:

- calculation of aircraft response accelerations (Pdot, Qdot, and Rdot) from high sampling rate response
- calculation of vertical and lateral accelerations at the aircraft c.g.
- data compression of the available flight parameters

The data compression techniques to be employed on-board the aircraft will eliminate most of the data processing procedures required for the original F-16 A/B FLR data reduction, including Quicklook and Loads Edit. The in-flight compressed data will retain the maneuver time history sequence provided by these procedures.

The peak-valley maneuver indicators for inflight data compression will include: the accelerations, Nz and Ny, roll response parameters, P and Pdot, and computed bending moment loads on the horizontal tails and the vertical tail. There are provisions to compress the ground time histories in much the same manner based on relevant ground maneuver parameters.

The AMU will contain sufficient memory to retain 10 or more flights of Type 3 data, depending on the flight maneuver activity. In the event that the memory capacity is reached, the oldest data in memory will be overwritten. The Type 3 data will be extracted from the AMU by the MLV unit during the same 100-hour phased inspection as the Type 2 data resulting in the capture of approximately 10 percent of flight activity on all F-16 C/D aircraft.

The compressed time histories resulting from the Type 3 data collection will be processed by revised loads prediction procedures for development of both airframe and landing gear operational loads spectra.

4.4 Engine Usage Data (Type 4)

The Type 4 data processed by the SAU and stored in the AMU with the Type 3 data will greatly increase the availa e e gine usage data over the methods employed in the original F-16 FMP. The engine usage parameters will be expanded to include true freestream air temperature, longit inal acceleration, turbine fan RPM, and fan inlet temperature as shown in Table 4.1.

The processing procedures for the Type 4 data will incorporate a floating aperture algorithm for data compression similar to that to be used for the Type 1 data processing. This will result in high resolution time histories of operational engine usage for subsequent use in the development of operational usage spectra for the engine. The Type 4 data will be stored in the AMU along with the Type 3 data and will be extracted at the same 100-hour phased inspection intervals as the Type 2 and Type 3 data.

4.5 CSFDR Incorporation

The incorporation of the CSFDR system on the USAF F-16 C/D aircraft will greatly enhance the capabilities for monitoring operational usage in addition to providing the information necessary for detailed and timely mishap investigation. Current plans anticipate that the system will be available in 1987. Consideration is also being given to the installation of the CSFDR system in other F-16 variations including a retrofit of the system into USAF F-16 A/B aircraft.

5. CONCLUSION

The Force Management Data Package provided to the United States Air Force for the F-16 aircraft by General Dynamics is fulfilling the objectives of Task IV of the Air Force Aircraft Structural Integrity Program. The data collection and data management procedures for the Air Force to successfully manage the structural maintenance program for the F-16 aircraft during operation of Task V are well established. The future F-16 Force Management Program will provide increased capabilities to monitor force operations and information for structural maintenance planning.

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US ARMY HELICOPTER OPERATIONAL FLIGHT LOADS

by

Donald J. Merkley
and
Harold K. Reddick, Jr.
Applied Technology Laboratory
US Army Research and Technology Laboratories (AVSCOM)
Fort Eustis, Virginia 23604

SUMMARY

The evaluation of fatigue critical components of the helicopter requires a well defined mission loading spectrum which directly depends upon the actual operational usage of the helicopter. The US Army began concerted efforts in 1964 to acquire realistic service usage data for their operational helicopters. Initial measurement programs were made on aircraft operating routinely within the US. These efforts were subsequently extended to include helicopters performing combat and support missions in Vietnam, cold weather operations in Alaska, and nap-of-the-earth (NOE) training missions at Fort Rucker. In addition, a comprehensive Operational Load Survey was conducted to acquire detailed knowledge of the rotor aerodynamic environments and structural dynamic response simultaneously. The Army's most recent effort in flight loads monitoring has been the development and demonstration of the Structural Integrity Recording System (SIRS) which includes an airborne microprocessor based recorder, a portable flight line retrieval unit, and a data processing package.

This paper highlights the findings from the Army's helicopter service usage programs and the Operational Loads Survey program and describes the development and capability of SIRS.

INTRODUCTION

In order to determine the useful life of any aircraft, or the fatigue life of any component, three factors must be known. They are: (1) fatigue strength, (2) fatigue loads, and (3) the frequency of occurrence of these loads. These factors, when used in conjunction with a cumulative damage fatigue theory, result in the prediction of the fatigue life for structural components. The fatigue strength of the structure is determined from laboratory testing. The fatigue loads that the structural components will be subjected to in service are determined a short time after construction of the helicopter by conducting a flight load survey. Here, the helicopter is heavily instrumented and is flown to the various flight regimes specified in the design mission spectrum. The corresponding loads for each regime are measured. The third major factor, loading frequency of occurrence, is established initially from a specified mission profile generated generally by the procuring agency or the contractor. The mission profile, which is sometimes referred to as a flight or mission spectrum, is defined in the ideal sense as:

> "The most representative history of ground and flight conditions that a given vehicle will encounter during its lifetime."

The ground and flight conditions encompass variations and combinations of aircraft physical parameters such as external configuration, weight, and center of gravity; flight parameters such as airspeed, altitude, rotor RPM, and engine torque; and any other parameters such as gust loading spectra, which affect the lifetime of the helicopter structural components. This specified profile may be developed from any of several data sources that include statistical surveys conducted on similar operational helicopters, discussions with pilots and helicopter users, and anticipated scenarios in which the system being developed will be used. The loading frequency of occurrence can only be determined once the helicopter is fielded. Because of the various, frequently changing roles that the helicopter is required to perform, it is this load frequency of occurrence spectrum that is oftentimes the key variable of uncertainty for establishing structural component fatigue lives.

Fatigue-related problems associated with the helicopter differ from those associated with fixed-wing aircraft, by virtue of the rotating wings that produce inherent

cyclic loads throughout all components of the system. Figure 1 shows a typical comparison of the stress histories for a transport helicopter blade root end and that for a typical commercial transport wing root. The helicopter dynamic components tend to be loaded at frequencies that are multiples of the rotor speed--typically 2 to 20 Hz. This loading represents 7 million to 70 million cycles per 1000 operating hours. Thus, these components experience a much higher frequency of cyclic loading, but with a narrower spread of the loading, than the fixed wing aircraft component. Both fixed-wing and rotary-wing components are subjected to ground-air-ground cyclic loading. The problem of fatigue and structural integrity has continually grown in importance as demands for extended helicopter service life, increased operational performance, and the severity of operating conditions have increased.

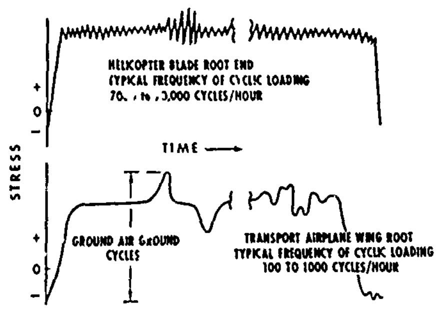

Figure 1. Typical Stress Spectra for Rotary-Wing and Fixed-Wing Aircraft.

The Applied Technology Laboratory (ATL), of the US Army Reasearch and Technology Laboratories, Aviation Systems Command (AVSCOM), has had a long history of measuring flight loads on operational helicopters. Recognizing an inadequacy in design criteria and specifications for the Army's aircraft, ATL launched a program in 1964 to obtain realistic mission and load spectra for the Army's operational aircraft. As a result, operational flight loads have been collected for several helicopter types in varied environments.

The oscillograph has been the primary and most successful means of recording these data. The SIRS has been developed for tracking the fatigue accumulation on helicopter structural components while at the same time providing useful information on the helicopter's operational usage. The SIRS provides a feasible flightworthy microprocessor system, capable of storing and processing large quantities of data in solid state memory with total automation.

This paper presents an overview of the US Army's work in operational flight loads technology, including highlights from helicopter service usage programs and the Operational Loads Survey program, and describes the development and capability of SIRS. Recommendations are made for needed future work and technology thrusts in this area.

REQUIREMENTS

The overall requirement for a comprehensive flight loads program lies in the need to be able to identify and record damaging flight regimes and to estimate how much fatigue damage has degraded component life during normal operations. This information can then be used to provide a safer environment and, in some instances, longer usable life of the aircraft and its subsystems. The identification of damaging flight regimes must rely on some means of flight measurements, the analyses of these measurements, and extended interpretation based on statistical practices.

Structural design criteria and military specifications that are peculiar to the varied missions performed by Army aircraft are necessary to assure procurement of aircraft that will satisfy the mission requirements of the Army and assure satisfactory service throughout their useful life. Historically, Army helicopters have been designed to mission profiles based on one or several sources that included existing military specifications such as MIL-S-8698 and AR-56, FAA specifications such as CAM-6, and contractor established specifications. The operational flight loads surveys conducted by ATL especially emphasized certain shortcomings of these specifications to represent the actual loads and flight spectra of the Army's helicopters in the field. The more recent procurements, the UH-60A (BLACK HAWK) and the AH-64 (APACHE), were designed to specifications specifically written for these aircraft, based on the flight loads data base established by ATL with the operational measurements, combined with expected mission profiles defined by the Army.

ATL APPROACH

The US Army has engaged in programs directed at the measurement and evaluation of the operational usage of its helicopters since 1964. The general objective of these programs has been to measure flight parameters and loads incurred by operational aircraft in order to better define helicopters' operational usage and associated flight load spectrum, thereby permitting improved design criteria and specifications, and a more reliable estimate of dynamic component fatigue life. In addition, the usage programs provide data for establishing the mission profile for a given type of aircraft and define the relationship of the recorded operating spectrum to that used in the initial design of the helicopter. Table 1 summarizes the usage programs. Initial measurements, documented in References 1 through 3, were made on aircraft operating

TABLE 1. US Army Helicopter Service Usage Measurement Programs

MEASUREMENT PROGRAM	NO. OF AIRCRAFT	FLIGHT HOURS
CH-47A		
CARGO AND TRANSPORT, USA	6	165
CARGO AND TRANSPORT, SEA	4	235
ARMED AND ARMORED, SEA	3	207
CH-54A		
CARGO AND TRANSPORT, USA	3	110
CARGO AND TRANSPORT, SEA	3	418
OH-6A		
COMBAT OBSERVATION, SEA	3	216
UH-1B/H		
COMBAT TRAINING, USA	4	219
COMBAT OPERATIONS, SEA	3	203
UTILITY OPERATIONS, ARCTIC	2	88
UTILITY OPERATIONS, ARCTIC	1	18
NAP OF-THE-EARTH TRAINING, USA	1	18
AH-1G/S		
COMBAT OPERATIONS, SEA	5	408
OPERATIONAL LOADS SURVEY, USA	1	28
SIRS EVALUATION, USA	10	337

65 66 67 68 69 70 71 72 73 74 75 76 77 78 79 80 81

within the Continental United States and were subsequently expanded to include helicopters performing combat missions in Southeast Asia (Vietnam), References 4 through 9, cold weather operations in Alaska, Reference 10, and NOE training at Fort Rucker, Alabama, USA, Reference 11. The flight loads investigations on helicopters in Southeast Asia (SEA) provided significant information of the operational usage of these helicopters in a combat environment and enabled comparisons with the same helicopters in a peacetime or simulated combat environment. Operational usage data have been collected for the following class-model helicopters: transport - CH-47A, crane - CH-54A, utility - UH-1H, attack - AH-1G, and observation - OH-6A. They are shown in Figure 2. The helicopters were instrumented and an oscillographic recording system was installed

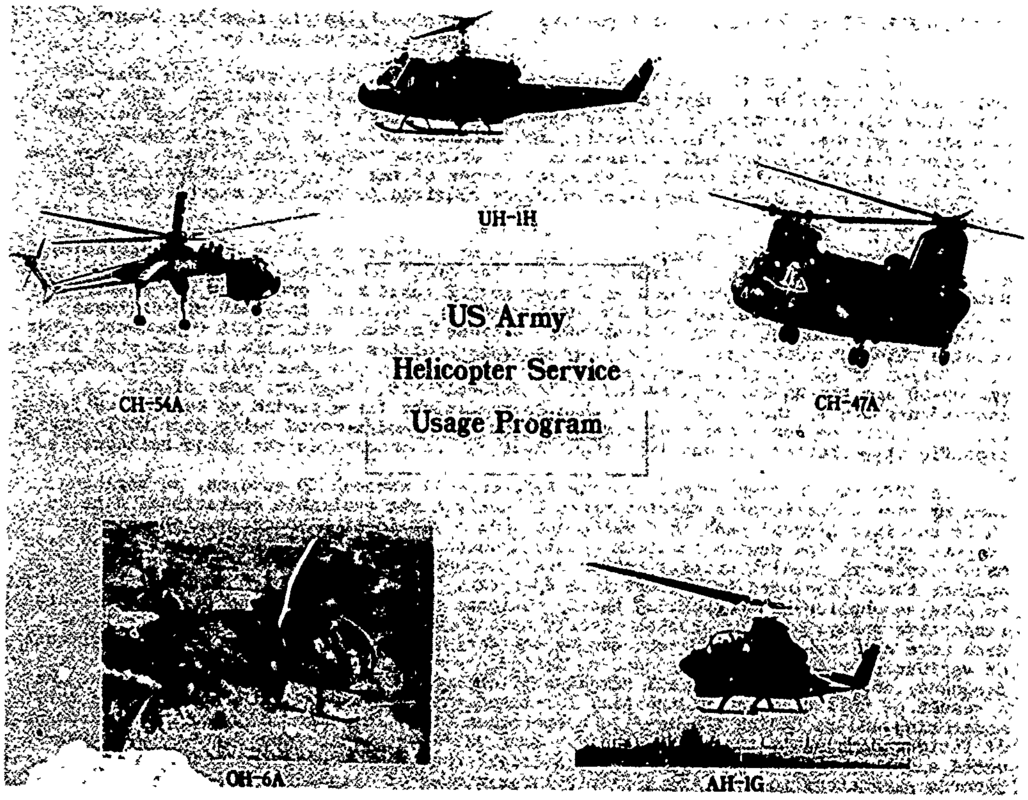

Figure 2. Operational Helicopters for Measurement Programs.

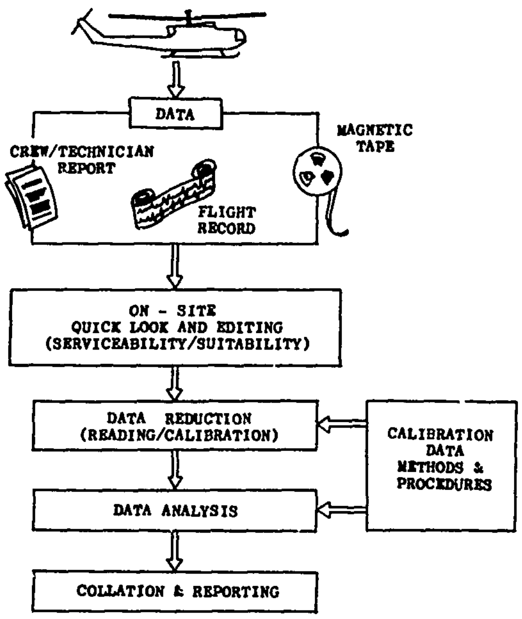

Figure 3. Data Recording and Reporting Process

to measure the parameters denoted in the Appendix. After each recorded flight, a field technician, aided by the pilot, filled out a special form to log the supplemental data needed to process the oscillogram data. Such additional information included: (1) the flight date, (2) helicopter configuration, (3) mission type, (4) airspeed and rotor rpm at certain check points, (5) takeoff elevation, (6) barametric pressure, (7) temperature, (8) base location, (9) time, (10) fuel weight, and (11) armament weight. Items 8-11 were recorded for both takeoff and landing conditions. In addition, the field technician logged the serial number for each transducer so that the calibration data could be correlated with the recorded data during final data processing. The data recording reduction process is depicted in Figure 3.

The AH-1G Operational Loads Survey (OLS), Reference 12, was not directed specifically at defining service usage, but rather entailed a comprehensive flight test program to acquire detailed knowledge of the rotor aerodynamic environments and the attendant structural dynamic responses simultaneously. In the process of these measurement programs, a large operational usage data base has been established, various methods of recording and techniques of analyzing the data have been evaluated, and the results have been applied to new analysis programs for improved prediction capability.

The measured data from the operational usage measurement programs were processed and analyzed according to distinct flight phases termed mission segments. Data were also reduced and presented in the form of time and occurrence tables, histograms, and exceedence curves. The tables give the time spent or number of parameter peaks distributed among the various combinations of parameter ranges; the histograms show the percentage of flight time spent in selected ranges of the flight parameters; and the exceedence curves indicate the number of hours required to reach or exceed specific parameters.

As operational flight loads data from the measurement programs became available, the studies in References 13 through 15 were performed to analyze these operational data, assessing how they differed as a function of theater of operation and mission assignments, how they varied from the design fatigue spectra, and what effect these variations have on the lives of helicopter fatigue critical structural components. The results which characterized the helicopter usage profiles were quite instrumental in the formulation of the mission profiles for the Army's new helicopter system developments, the UH-60 and the AH-64, as attested to in Reference 16. However, most of the assessments within these studies on changes in fatigue lives tended to be qualitative, e.g., "it must be concluded that components on the armed/armored helicopter would have a significantly shorter fatigue life than identical components on the cargo helicopters."

In 1972 the US Army contracted the Reference 17 through 20 studies, authorizing each of four US helicopter manufacturers to analyze the available operational data on their specific aircraft and to develop and update utilization spectrum with which to compare against the design spectrum. This updated spectrum was ultimately used by each manufacturer to predict critical structural component fatigue lives for comparison against those predicted by the original spectrum. Each manufacturer found it necessary to make assumptions in converting the operational data to a format appropriate for his respective analysis. The studies clearly demonstrated that the measured operational data were not amenable for direct use by the fatigue analyst in determining the effects of various operational environments on fatigue-critical components, because the frequency and sensitivity of all damaging flight conditions were not identifiable. As a result of this conclusion, the Army initiated the Reference 21 study of methods for monitoring and recording the in-flight operation of helicopters for specific assessment of fatigue damages to their structures. Four in-flight monitoring systems were evaluated and the recommended approach was a flight condition monitoring method which records parameters to identify a fairly large number of flight conditions upon which fatigue damage assessment is based. Development of a flight condition monitoring method, or SIRS as it is termed, began in 1975 and has been used to monitor service usage of the G and S models of the AH-1 helicopter.

USAGE MEASUREMENT PROGRAM RESULTS

Flight profile data from each of the operational usage surveys categorize the total flight time into four basic mission segments: (1) takeoff and ascent; (2) maneuvering; (3) descent, flare, and landing; and (4) steady state. These mission segments are defined in Reference 5 as follows:

"During the first three mission segments which comprise the transient part of flight, the stick position traces seem to deviate, while the airspeed and altitude traces manifest frequent changes. Mission Segment 1 (takeoff and ascent) includes not only the takeoff and climb to the initial steady-state altitude but also the steady ascents to other steady-state altitudes. Mission Segment 2 (maneuvering) consists of any transient parts of the flight which are not characteristic of Mission Segments 1 and 3. During maneuvering, the normal acceleration trace is usally very active. In addition to the unsteady part of flare and landing, Mission Segment 3 (descent, flare and landing) includes the unsteady part of any descent whether intended for a new steady flight altitude or for landing. Mission Segment 4 (steady state) includes those parts of the flight where the stick traces are relatively steady and where the airspeed and altitude traces are steady or changing smoothly. Such characteristics prevail during cruise, hover, steady ascent, and descent."

Oftentimes the usage measurement data trends for key mission segments and flight parameters deviate from those specified or developed for the design spectrum. Caution must be exercised, however, in reaching conclusions on component fatigue life changes based only on certain segment or parametric variations--all variations must be considered. This is best illustrated by the Reference 18 study wherein select dynamic component fatigue lives for the AH-1G helicopter were calculated using a mission profile based on SEA operations and compared with those determined using the original frequency of occurrence spectrum. The findings were:

"Although there were some relatively large modifications in some areas of the spectrum, the net effect on the resulting fatigue life was not significant. This is attributed to the fact that even though some of the individual changes seemed rather drastic, they tended to compensate for each other. The changes in mission segments, wherein the time spent in level flight was reduced and the time spent in maneuver was increased, would lead one to expect a reduction in fatigue life. However, this was apparently compensated for by the reduction in severity of the airspeed distribution, since the oscillatory loads are strongly dependent upon airspeed."

A short summary of each of these major measurement programs is presented in the following sections, catagorized by aircraft.

CH-47A Chinook

The twin-tandem CH-47A Chinook helicopter was designed to meet the US Army medium lift requirements as a personnel transport and cargo carrier. The helicopter is a twin turbine engine, tandem rotor medium-lift aircraft with a design gross weight of 28,500 pounds. Cargo can be transported either internally or externally. Flight service usage evaluation programs have been conducted on the CH-47A flown in three different mission assignments. Two of the missions were flown in SEA under actual combat conditions, one as an armed/armored helicopter, Reference 4, and one as a cargo/transport helicopter, Reference 5. The third mission was flown as a cargo/transport helicopter during simulated maneuvers in the United States, Reference 3.

Figure 4 presents, for each of the three CH-47A measurement programs, the percentage of the total time in each mission segment and compares it against the profile from CAM-6, Reference 22, and the contractor's original design profile for the CH-47A, Reference 24. The mission segment breakouts for the two cargo/transport helicopters are generally in close agreement. Variations in the maneuver and descent segments may be due to the normal scatter that can be expected with this type data, or could possibly be due to variations in factors associated with operating in friendly and hostile environments.

The mission segment breakout for the armed/armored CH-47A operating in SEA varies considerably from those obtained for the cargo/transport helicopters. The high percentage of time spent in the maneuver segment, which is essentially unsteady forward flight at fairly constant altitudes, reflects the unsteady nature of the gunship's mission assignment in supporting ground operations.

Comparison of the empirical mission profiles based on CAM-6 and the CH-47A design spectra shows higher steady-state time percentages than are revealed from flight-measured data. The percentage of time in ascent and descent is consistantly higher for all three flight measured programs. The design profiles show significantly lower maneuver time percentages than measured for the armed/armored mission helicopter.

Cumulative frequency distributions for airspeed, gross weight, altitude, and vertical load factors based on measured data from each of the three service measurement programs is presented in Figure 5. Airspeed distributions are expressed as a percentage of V_A, where V_A is defined as the maximum attainable level flight velocity considering gross weight, usable power, blade stall, and structural limitations. Comparison of the cumulative airspeed frequency distributions shows that the SEA armed/armored and the USA

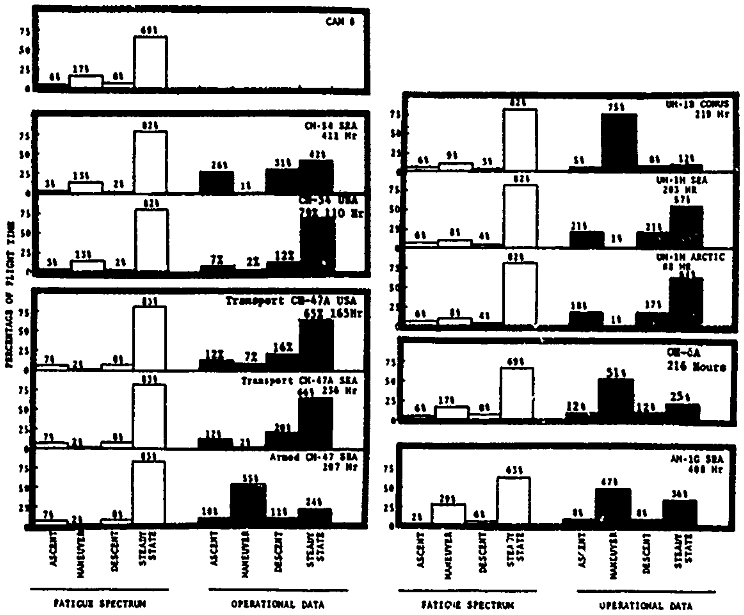

Figure 4. Design and Operational Mission Segment Comparisons

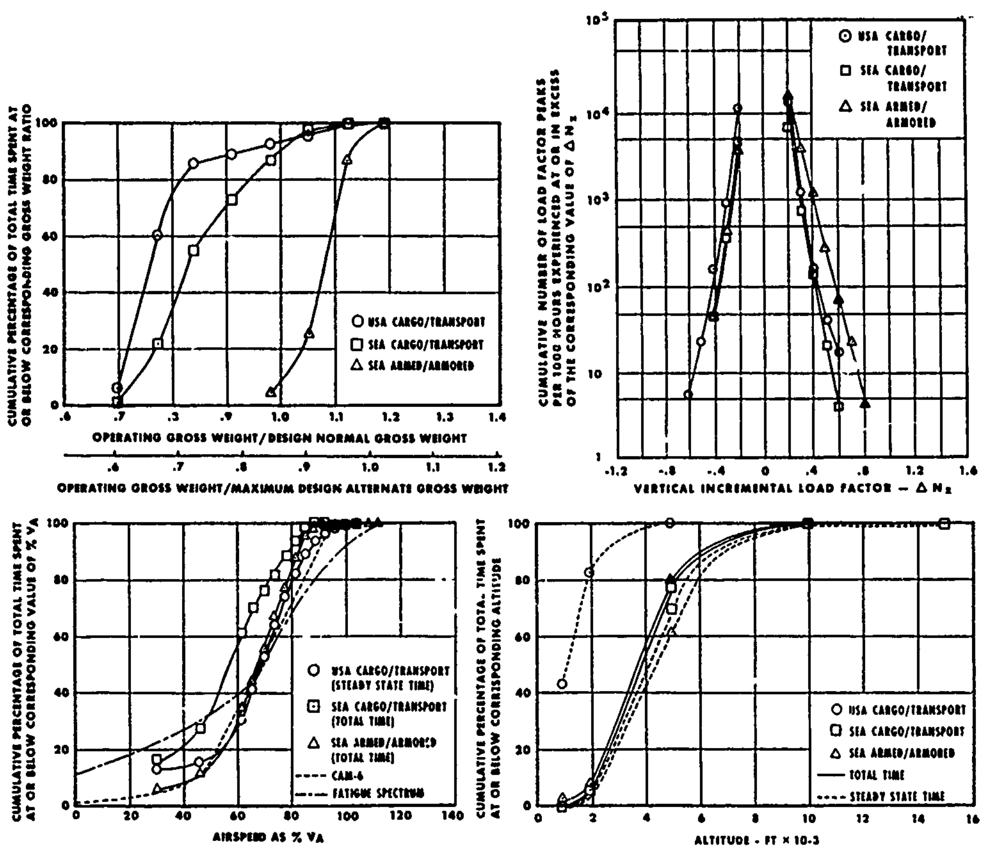

Figure 5. CH-47A Cumulative Frequency Distributions for Gross Weight, Load Factor Airspeed and Altitude

cargo transport helicopters spent greater percentages of time at higher airspeeds than the SEA cargo/transport helicopter. The variations between the USA and the SEA cargo/transport aircraft may be influenced by geographic and climatic conditions as well as by the change from a friendly to a hostile environment. The cargo/transport helicopters spent more time at the lower airspeeds than the SEA armored/armored helicopters. Finally, the maximum airspeed attained by the SEA armed/armored helicopter was higher than that attained by either of the cargo/transport helicopters. A comparison of the flight-measured airspeed spectra with the CAM-6 spectrum reveals that the CAM-6 spectrum is in fairly close agreement with the data for the USA cargo/transport and the SEA armed/armored helicopters. The fatigue spectrum, upon which the CH-47A was originally designed, is not in very good agreement with flight-measured data.

Little similarity is noted in the cumulative gross weight frequency distributions for the three CH-47A mission assignments, particularly between the cargo/transport and the armed/armored helicopters. The SEA armed/armored helicopter spent 90 percent of the total time at gross weights in excess of the 28,500 design normal gross weight, whereas the SEA and USA cargo/transport helicopters spent 10 percent and 7 percent, respectively, of the total time above this value. The specialized nature of the SEA armed/armored helicopter's mission requires it to carry a considerable amount of attached armament and armor to and from the target area. The weight variations occurring during the mission would be due only to fuel, oil, and ammunition expended. The cargo/transport helicopter, however, would be loaded with troops or cargo either on its way to or returning from the target area, but usually not in both directions. Thus, the percentage of time spent by the cargo/transport aircraft at the heavier gross weight would be expected to be considerably less than those spent by the armed/armored aircraft. Differences in the cumulative gross weight frequency distributions for the two transport/cargo helicopters may be due, in part, to operational environments. Possibly, operations in a friendly environment would not be as efficiently performed as in an unfriendly one, or the need to carry higher payloads in the USA cargo/transport helicopter would not be as urgent as it would be in the SEA cargo/transport helicopter.

The cumulative altitude frequency distributions show that the two helicopters operating in SEA generally flew at higher density altitudes than did the cargo/transport helicopter flying in the USA. This is primarily due to the higher elevations of the flight terrain and the relatively higher temperatures prevalent in the area of operation in SEA. Conjecturally, another contributor could be that helicopters flying over terrain where enemy ground fire may be expected would fly at higher altitudes to avoid this danger.

The flight load spectra data in Figure 5 for vertical load factors are expressed as "at or above" (rather than "at or below") cumulative frequency distributions because the occurrences were summed cumulatively starting at the largest absolute value of load factor. The vertical load factor distribution for the three CH-47A mission assignments show a degree of variation of the positive vertical load factor data, particularly between the two cargo/transport and armed/armored aircraft. The negative load factor data are quite uniform. The higher vertical load factors and the higher frequencies of occurrence experienced by the armed helicopter are not surprising, and it would also be anticipated for this aircraft would be flown in a more erratic manner. It would also be expected that the frequency of occurrence of the higher negative load factor peaks would be greater for the armed aircraft than for the cargo aircraft. This, however, is not evidenced by the data available.

Comparison made in Reference 14 of the frequency of occurrence of maneuver and gust-induced vertical load factors revealed that the gust-induced vertical load factors encountered by a helicopter are only a small percentage of the total load factor experience. For example, the number of maneuver induced vertical load factor peaks experienced per thousand hours at $\Delta n_z = .4g$ or greater is approximately 1010 for the SEA armed helicopter. The number of gust-induced vertical load factor peaks is 4, or .39 percent of the total factor experience. Vertical load factor-airspeed trends examined in Reference 14 show that for both the cargo and armed helicopters load factors were most frequently encountered in the 65-knot to 100-knot airspeed range, rather than at the lower or higher airspeeds.

Reference 19 cites the applicability of this operational usage data in the CH-47 development:

"Operational use of the CH-47A as compared to the design mission profile led directly to the rework of the forward and aft rotor shafts and contributed to reduced retirement lives for several dynamic system components. Exceedence of the airspeed limitations of the operator's manual is cited as the principal cause of the reduced fatigue lives. The experience gained in evaluating the operational use of the CH-47A helicopter was used in the development of the CH-47B and CH-47C models. The structural performance of the growth models shows a great improvement over the CH-47A, at least a part of which should be attributed to the CH-47A operational experience. No operational survey has been conducted on the B and C models however, so the adequacy of their mission profiles cannot be evaluated."

CH-54A Tarhe

The CH-54A helicopter is a twin-turbine engine, six-bladed single-rotor, heavy-lift aircraft having a design normal gross weight of 38,000 pounds and a design maximum gross weight of 42,000 pounds. The helicopter is configured to carry heavy, outsized payloads or special purpose vans or pods from either a single-point or four-point suspension system. Twenty thousand pounds can be carried with the winch locked at a selected cable length. The four-point system uses four 6000-pound-capacity hoists mounted at hard points on the side of the fuselage. Helicopter crew consists of a pilot, a copilot, and an aft-facing hoist operator.

The operational data base for the CH-54A resulted from two measurement programs - one during simulated combat maneuvers at Fort Benning, Georgia, USA, Reference 2, and another during actual combat missions in SEA, Reference 7.

Figure 4 presents design and operational mission segment data as bar charts showing the percentages of time spent in each of the four mission segments: ascent, maneuver, descent and steady-state. The individuality of flight spectra and the importance of mission assignment in establishing the character of flight spectra become apparent when comparing the SEA and USA operational data. Only the percentages of time spent in the maneuver segment are comparable. For the other segments, the percentages of time corresponding to the actual combat mission versus the simulated combat mission are 42 percent versus 79 percent for steady state, 26 percent versus 7 percent for ascent, and 31 percent versus 12 percent for descent respectively. In this case the predicted simulated combat mission assignment flown in the USA was not indicative of the experience that prevailed in actual combat. Comparison of the flight measured mission segments with the emperical design mission profile reveals fair correlation between the data from USA operations and the design profiles. However, the data from SEA operations shows poor correlation with the design profile: the steady-state percentages from operational data is only approximately half that of the empirical profiles and the ascent and descent segments from operational data are considerably higher the respective segments of the empirical profile.

The flight spectra data for the SEA operations were divided into samples of 203 and 207 hours to examine the effects of sample size on the resulting flight spectrum. The percent deviations from the mean for the two data samples are ±.12 percent for ascent, ±.15 percent for maneuvers, ±3.1 percent for descent, and ±2.9 percent for steady state. These deviations are not considered excessive, in that the combat situation, and thus the flight spectra, encountered from day to day or over a period of time could vary significantly due to changes in objectives or tactics, or fluctuations in weather conditions.

Cumulative frequency distributions for airspeed, gross weight, altitude, rotor speed (RPM), and vertical load factor, based on measured data from the usage measurement programs in the USA and SEA, are presented in Figure 6. Analysis of this data is presented in the following paragraphs.

The cumulative airspeed frequency distributions show that the CH-54As in the SEA theater spent 45 percent of their time at airspeeds between 67 and 100 percent V_A, wherein those in the USA spent over 55 percent of their time in this airspeed range. The distributions for the two SEA samples showed close agreement. The SEA flight measured airspeed spectrum compares quite well with the CAM-6 spectrum over the major portion of the airspeed range, deviating somewhat at the low and high airspeeds. The CAM-6 spectrum underestimates the percentage of total mission time at airspeeds below the 50-percent V_A value and overestimates the time spent above the 90 percent V_A value. The USA flight measured airspeed spectrum does not show as close correlation with that from CAM-6. The CH-54A fatigue spectrum, on the other hand, does not compare as favorably with either the USA or SEA flight measured data, showing only reasonable correlation with the SEA operational data at high airspeeds above 80 percent V_A, and the USA operational data possibly at low airspeeds below 30 percent V_A.

The cumulative frequency distribution for gross weight showed the SEA helicopters to spend a greater percentage of time at lower gross weights. At the high gross weights, the SEA and USA helicopters spent 10 percent of the time in excess of the design normal value of 38,000 pounds. The USA helicopters were operated at higher weights. The difference in overall distributions could be due in part to operational environments and mission assignment. The exceedance in design normal values would be expected to be just the opposite of that shown, in that it would seem as though for the CH-54A operating in a friendly environment the transport of cargo in excess of design values would not be done as frequently as for the CH-54A in a combat environment.

The cumulative rotor speed frequency distribution developed from the rotor RPM histograms for the SEA CH-54As shows that the spread of normalized rotor speed was .97 to 1.05 of the 100-percent rotor RPM value. Figure 6 shows that the large majority of total mission time, 76 percent, is spent above the 110 normalized rotor speed value. The CH-54A flight manual specifies that operation at rotor speeds between 100 and 104 percent is permitted but not on a continuous basis. Thus, the operating rotor speed data shows that during large portions of the total mission time both aircraft were operated at rotor speeds exceeding the limits recommended in the flight manual.

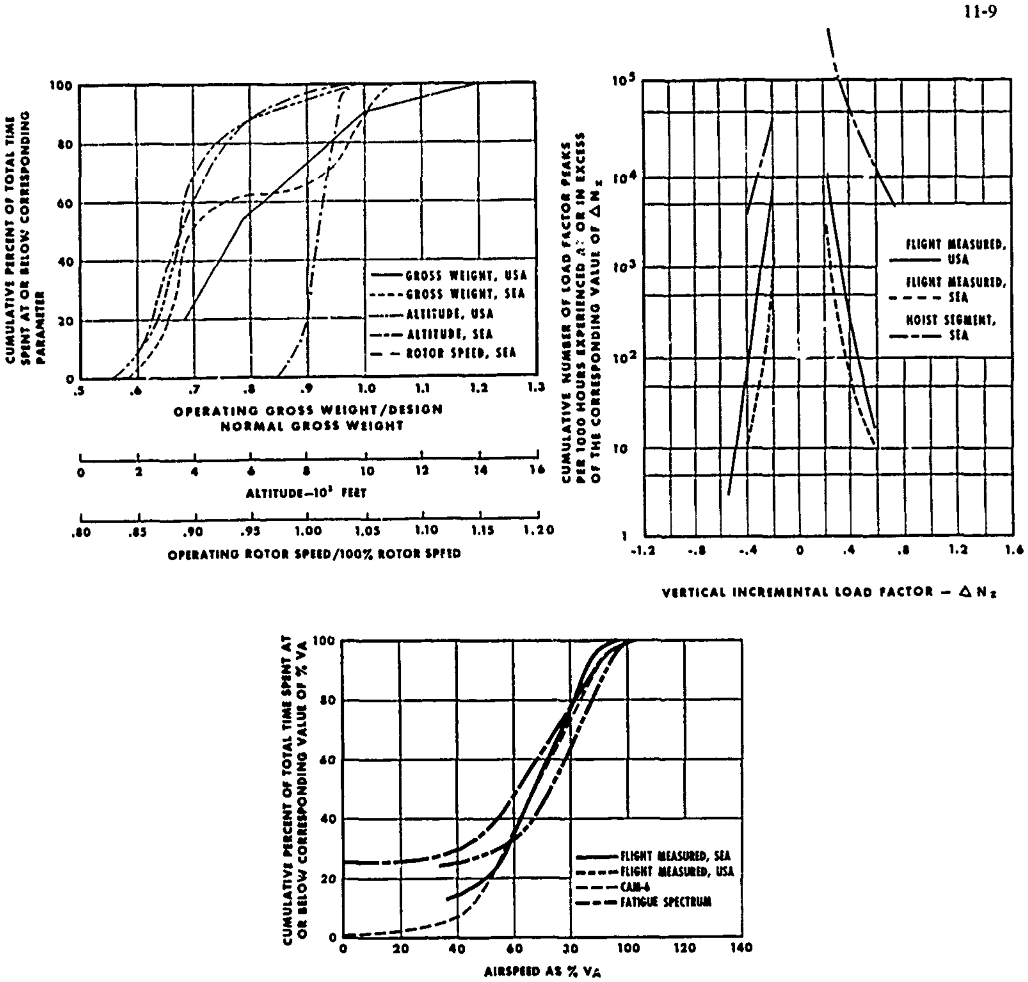

Figure 6. CH-54A Cumulative Frequency Distributions for Gross Weight, Altitude, Rotor Speed, Vertical Load Factor, and Airspeed.

Altitude distributions for the CH-54A in SEA showed that although flight altitudes as high as 10,000 feet were experienced, less than 5 percent of the total mission time was spent above 8000 feet. About 25 percent of the total time was spent at altitudes between 4000 and 8000 feet, and 60 percent of the time was spent between 2000 and 4000 feet. Altitudes below 2000 feet accounted for less than 9 percent of the mission time. Trends for USA operations were similar. Comparison of the altitude distribution of the CH-54A to those of the AH-1G and OH-6A operating in SEA shows the CH-54A to have a consistently smaller cumulative percentage of total time from 1000 to 10,000 feet. This indicates that the CH-54A spent a greater percentage of its time at higher altitudes. This is likely due in large part to the mission differences between the aircraft. Also, flying over terrain where enemy ground fire might be expected, the CH-54A would normally cruise at higher altitudes to avoid this danger.

The vertical load factor frequency distributions for the USA and SEA mission assignments show a degree of variation for both positive and negative values. Higher vertical load factors and higher associated frequency of occurrence were experienced by the CH-54A in the USA theater. For both the SEA and USA missions, the incident and magnitude of positive vertical load factor peaks are higher than for the negative ones. The data shown includes both gust- and maneuver-induced load factors. However, the gust-induced vertical load factor peaks did not exceed ±.3g and their frequency is a factor of 20 less than maneuver induced. In addition to the gust- and maneuver-induced vertical load factors, a hoist segment was added to SEA operations to study the accelerations encountered by the helicopter during either a cargo pickup or cargo drop while the aircraft were hovering. A high incident of vertical load factor peaks were measured in hoist operations. The data should be used with caution, however, because the period of the data sample is small. The data, nonetheless, does tend to suggest that higher vertical load factor peaks were experienced during hoisting than during normal flight maneuvers. The incidences of vertical load factor peaks encountered within a given airspeed range were investigated for the CH-54A in SEA. The following trends were

noted: the greatest number of positive vertical load factor peaks of Δn_z = +.3g or greater occurred within an airspeed range of 0 to 40 knots and 70 to 100 knots; only a small number of lower magnitude vertical load factor peaks occurred above 100 percent V_A; the maximum vertical load factor peak did not exceed Δn_z = +1.0g or -0.5g.

UH-1H Iroquois

The UH-1H "Huey" is an all-metal, single-engine helicopter having a design maximum gross weight of 9500 pounds. A single, two-bladed, semi-rigid teetering main rotor provides lift, and a two-bladed, semi-rigid, delta-hinged tail rotor provides antitorque and directional control.

A large data base has been acquired on the UH-1H helicopter in various programs, including training activities for combat assault and NOE in the USA, actual combat missons in Southeast Asia, and cold weather operations in the Arctic. The early combat training in the USA was actually conducted with a UH-1B, but will be included here for discussion purposes. Four basic missions were flown in SEA: combat assault, direct combat support, command and control, and passenger transport. Only the combat assault missions can be compared to the similar measurement programs in SEA for the AH-1G, armed CH-47A, and the UH-1B since those misions include operations in a hostile environment. The other missions of the UH-1H in SEA were primarily non-hostile operations consisting predominately of resupply and personnel transportation. The percentages of the 249 missions flown in SEA are as follows: combat assault, 16.5 percent; direct combat support, 71.1 percent; command and control, 8.4 percent; and passenger transport, 4.0 percent.

Multi-channel flight data were recorded on two UH-1H helicopters operating from Fort Greely, Alaska. Data were processed and analyzed by two different techniques, the Four Mission Segment technique and the Flight Condition Recognition (FCR) technique. The Four Mission Segment technique is the same as that used to process the helicopter operational usage data in Southeast Asia. The FCR technique processed the data according to the occurrence of 20 different flight conditions within seven mission segment· (1) ground operation, (2) hover, (3) ascent, (4) level flight, (5) descent, (6) tra tion and (7) autorotation. The FCR technique, which is basically the FCM concept discussed in the section on monitoring systems for fatigue damage, provided more detailed results and better resolution of the operational usage than did the Four Mission Segment technique. Significant fatigue-damage manuevers were more easily identified, and the manuever-induced normal load factors and their duration were better defined. A comparison of the Arctic UH-1H data with the Southeast Asia UH-1H data, both processed using the Four Mission Segment technique, showed that the Artic data had greater amounts of time at higher values of airspeed, gross weight and engine torque, and lesser amounts of time at equivalent rates of climb and descent.

Figure 4 is a comparison of the operational data and fatigue spectra for the UH-1B USA, and the UH-1H SEA and Arctic measurement programs. This figure indicates very poor correlation with the design fatigue spectrum obtained from Appendix A of CAM-6, Reference 22. The greater percentage of maneuvering flight for the UH-1B USA measurements may be attributed to the training nature of the flights. For example, there were 58 practice autorotational landings performed. On the basis of this comparison, it appears that these gathered data further demonstrate the individuality of flight spectrum data and the importance of mission assignment in establishing the characteristics of the operational usage spectrum.

Figure 7 compares the cumulative airspeed frequency distribution for the Arctic and Southeast Asia UH-1H measured data with those for the CAM-6 spectrum and the UH-1H design fatigue spectrum. Except for the higher Arctic data curve at higher airspeeds, the two curves have generally the same shape. Below 75 percent V_A, the measured Arctic data is in fair agreement with the two spectra curves. However, for the Southeast Asia data below 85 percent V_A, the agreement is poor. If it is assumed that most of the fatigue damage occurs at higher airspeeds, then the CAM-6 and UH-1H design fatigue spectra are not necessarily conservative with regard to this parameter.

The comparison of the cumulative gross weight frequency distribution for the Arctic UH-1H data and the SEA UH-1H data is also shown in Figure 7. The shape of the two curves closely agree. The Arctic data does indicate that about 4 percent of the flight time was spent at gross weights above the design maximum. This high gross weight operation may be attributed to the greater power available at the low ambient temperatures and special mission requirements.

To investigate the effects of cold-weather operations on the loads of various dynamic helicopter components, the component loads were plotted against main rotor tip Mach number in Reference 10. An example plot is provided in Figure 7. This figure indicates that the main rotor blade oscillatory bending moments in the cold weather (-30°C) are much higher than those of the warm weather data (20°C). The beamwise and chordwise oscillating moments of the main rotor blade increased greatly as the main rotor tip Mach number increased. The high main rotor tip Mach numbers were a direct result of the extremely cold environment. The increased oscillatory moments can be explained by the aerodynamic compressibility effects on the rotating blade section, resulting in earlier separation of flow and a change in center of pressure.

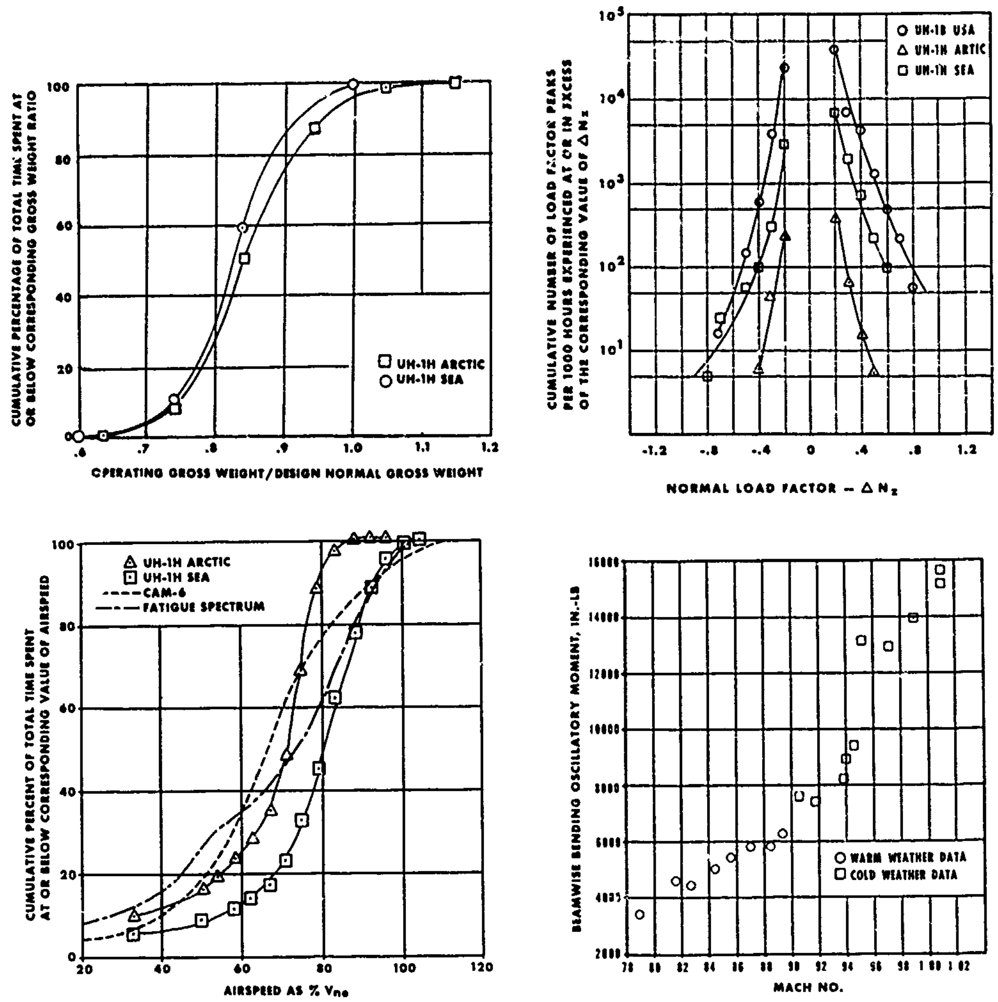

Figure 7. UH-1H Cumulative Frequency Distributions for Gross Weight, Normal Load Factor and Airspeed, and Blade Moment Versus Mach Number.

A UH-1H helicopter assigned to NOE training missions at Fort Rucker, Alabama, was instrumented with an oscillograph recording system to update the usage spectrum in the NOE flight environment. The data were processed by the FCR technique. Since helicopters during NOE missions perform many transient manuevers, some parameters were processed as peaking parameters (parameter values exceeding a given threshold) and others as excursions (specific parameter value changes, either increasing or decreasing, within a specific time span). Exhaust gas temperature, rotor speed, and vertical and longitudinal accelerations at the aircraft cg and the tailboom were processed as peaking parameters, while forward airspeed and engine torque were processed as excursions.

Additional parameters are required to identify the slower speed manuevers typical of those used in NOE missions. For example, pitch attitude aids the identification of quick stops, and yaw attitude is a good indicator of pedal turns in hover. Terrain clearance is also helpful in identifying pop-ups and bob-ups.

In general, it was found that the operational usage spectrum for the NOE mission program flown in training at Fort Rucker is relatively mild in terms of Nz data when compared with those derived from other measurement programs with a more conventional mixture of mission types.

AH-1G Cobra

The AH-1G "Huey Cobra" helicopter is a highly maneuverable, high-speed, single-engine gun ship. It has a design gross weight of 9500 pounds. Deployed as a ground-support weapons platform, the AH-1G has a controllable nose turret and two external store pylons. The crew consists of a pilot and a copilot/gunner.

Three AH-1G helicopters were instrumented and operated from bases in SEA to perform a combat-zone flight loads program. Data were recorded, processed, and analyzed following the Four Mission Segment technique. These data are reported in Reference 6 and indicate the time spent in the mission segments and parameter ranges; the number of peak parameter values occurring in the ranges of the given parameter, during each of the mission segments, and in the ranges of one or more related parameters; and the time to reach or exceed given maneuver and gust normal load factors. 408 valid data hours were recorded and separated chronologically into two data sets of about 200 hours each to test the validity of 200 hours as an adequate data sample. The differences between the two samples were minor, and the two samples were observed to be similar in their distributions of flight data.

The favorable comparison of the two sample sets is evidenced by the similar contrasts of these percentages with those previously prescribed as a guide for the expected distributions of the AH-1G flight time among the four mission segments. Figure 4 shows the comparison of the operational data with the fatigue sprectra. The largest discrepancy is in the maneuver and steady-state segments. The differences are ascribed to the fact that the anticipated sustained cruise to and from the target area was higher than that actually measured in the data of most of the flights. The actual combat data revealed that the AH-1Gs, while enroute to and from target areas, frequently left cruise altitudes with abrupt ascents and descents for apparent searches of enemy activity. Some of these departures were weapon passes; others were sufficiently pronounced in parameter changes to be also classified in the maneuver segment.

The flight measured airspeed spectrum for the AH-1G helicopter is shown in Figure 8, from Reference 15, with the corresponding fatigue spectrum data used to estimate

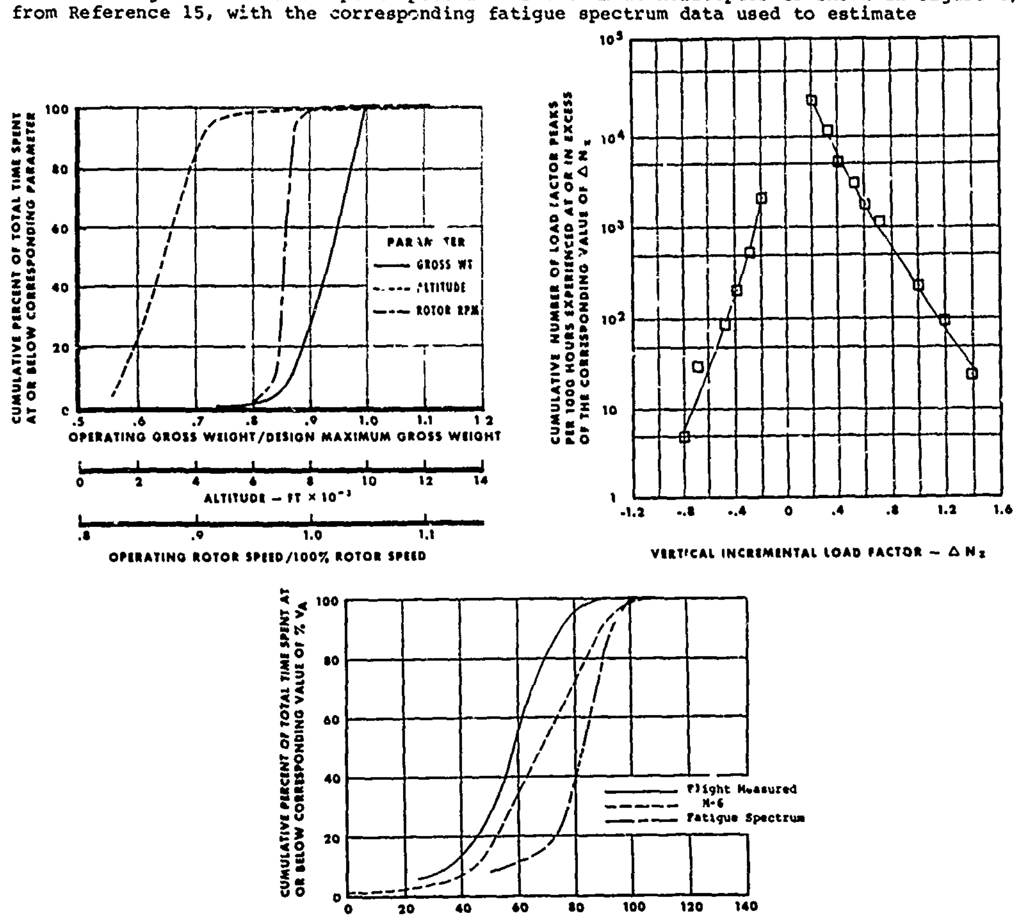

Figure 8. AH-1G Cumulative Frequency Distributions for Gross Weight, Altitude, Rotor Speed, Vertical Load Factor and Airspeed.

preliminary service lives and the airspeed portion of the CAM-6 spectrum. This comparison shows that agreement is not very good. The flight measured data shows considerably higher percentages of total mission time spent at low airspeed values than predicted by either of the empirical spectra. If it is assumed that the majority of component fatigue damage occurs at higher airspeeds, the AH-1G fatigue and CAM-6 spectra are conservative since both predict higher percentages of time at the high airspeed values than were actually recorded.

Flight altitudes as high as 15,000 feet were recorded, although the percentage of mission time spent above 10,000 feet was insignificant. Over 99 percent of the AH-1G total mission time was spent at altitudes of 8000 feet or less while 82 percent of the time was spent at altitudes between 1000 and 4000 feet.

The gross weight frequency distribution shows that the slope of the total mission curve is essentially constant at gross weight ratios greater than 0.875. This indicates that the percentage of time spent within gross weight ratio intervals of equal magnitude did not vary throughout the range of values 0.875 to 1.0. The total range of normalized rotor speed values was between 0.95 and 1.0 for over 95 percent of the total mission time.

OH-6A Cayuse

The OH-6A is an all-metal, single-engine helicopter. It is powered by a turbine engine driving a four-bladed main rotor and a tail-mounted antitorque rotor through a two-stage, speed reduction transmission. The aircraft is equipped with shock-absorbing landing skids. It has a design gross weight of 2400 pounds. Primarily an observation aircraft, it is capable of carrying pilot and three passengers, cargo, or armament subsystem. Two major configurations were observed during the recording period in SEA: the "lead ship", identified by a pilot and two gunners each with an M 60 machine gun, and the "wing ship", identified by a pilot and one gunner with an XM 27 minigun mounted on the left side. The Four Mission segment technique was used to analyze the 216 hours of data that were recorded in SEA.

Figure 4 presents the percentage of the total time in each mission segment and compares it against the contractor's original design profile for the OH-6A which was CAM-6. The primary differences are in the maneuver and steady-state segments. Only the ascent and descent segments show any degree of correlation. Comparing the OH-6A flight measured distribution with that obtained for the AH-1G, it is seen that they are very similar. Both the AH-1G and the OH-6A are armed, relatively lightweight helicopters used in ground support operations during combat. It would therefore be expected that the character of their flight spectra, in general, would be similar. The highest percentages of the total time were spent in the maneuver segment and vary only by approximately 3 percent between the two ships. Variations in the ascent and descent segments are approximately 4 percent, while the variation in the steady-state segment for the AH-1G and the OH-6A suggests that even though the general flight spectra characteristics are similar, they exhibit sufficient individuality to preclude using the flight spectrum of one to determine the component fatigue lives of the other. Since the flight loads associated with maneuvers are, in general, higher than those associated with steady-state flying, calculated component service lives established from each of these two helicopters could be higher than field experience justifies.

Figure 9 presents flight measured cumulative frequency distributions for airspeed, gross weight, altitude, rotor speed, torque, and vertical load factor for the OH-6A. The OH-6A flight measured airspeed data do not correlate closely with the respective fatigue spectra. The flight measured data show considerably higher percentages of total mission time spent at the low airspeed values than predicted by the empirical spectra. This same trend occurred with the AH-1G SEA flight measured data. If it is assumed that the majority of component fatigue damage occurs at the higher airspeeds, then the AH-1G and OH-6A design fatigue spectra should be conservative with regard to the airspeed parameter.

Over 27 percent of the OH-6A total mission time is above the design normal gross weight; however, very little time is at gross weights in excess of the design maximum value.

Flight altitudes as high as 10,000 feet were recorded, but the percentages of total mission time spent above 8,000 feet were insignificant. The bulk of the total time, 92 percent, was spent at the lower altitudes between 1,000 and 4,000 feet.

The range of OH-6A normalized rotor speed values was between 0.98 and 1.05 for over 95 percent of the total mission time. A large majority of the total mission time, 97 percent, is spent above the 1.0 normalized rotor speed value. The OH-6A flight manual states that continuous operation at rotor speeds up to 103 percent is permitted. From the figure it is seen that the OH-6A was operated at rotor speeds above the 103 percent value during 40 percent of the total mission time. Thus, the aircraft was operated during a significant percentage of its total mission time at rotor speeds exceeding the flight manual recommended limits.

Engine torque values were concentrated toward the high end of the torque range. Values up to 113 percent of maximum were experienced for very short periods; however, 73 percent of the total time was spent at torque values from 60 to 90 percent of the maximum allowable.

The total incident of positive and negative load factors is approximately equal. As with other helicopters, the incident of gust-induced vertical load factor peaks is much smaller than it is for the maneuver induced vertical load factor peak and has little influence on the exceedance curve characteristics. Although not shown in the figure, the highest positive vertical load factor peaks were experienced at airspeeds from 70 to 75 knots (60-64 percent V_A); the highest negative vertical peaks were experienced at airspeeds from 60 to 85 knots. Comparisons have been made to vertical load factor spectra from a survey made on the OH-6A during simulated combat operations in the USA. The OH-6A in SEA experienced both higher incidences and higher magnitudes of positive vertical load factor peaks and a lower incidence of negative vertical load factors peaks than OH-6A in USA operations.

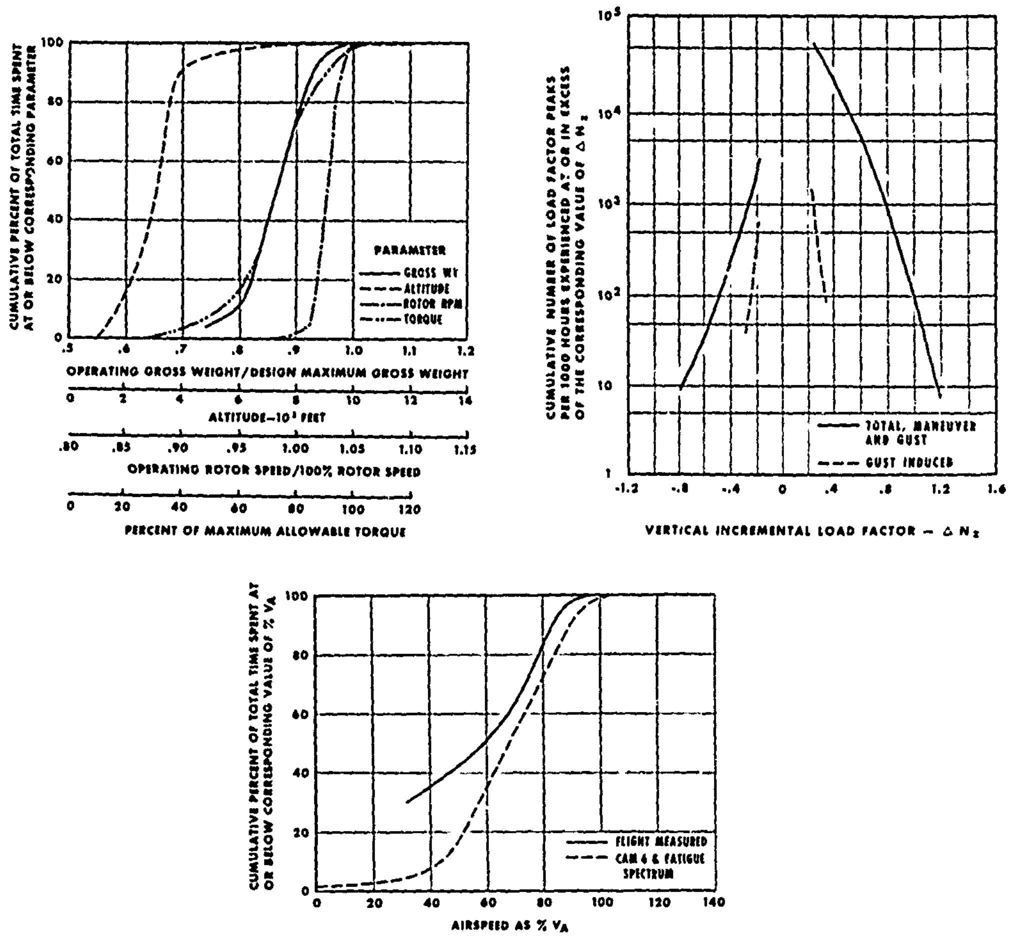

Figure 9. OH-6A Cumulative Frequency Distribution for Gross Weight, Altitude, Rotor Speed, Torque, Vertical Load Factor and Airspeed

AH-1G OPERATIONAL LOADS SURVEY (OLS)

The Operational Loads Survey (OLS) was unique as a flight loads survey and quite different from all the others. The intent of the program was not only to measure loads of critical components and count the number of flight parameter occurrences but to gain experimental insight of rotor aerodynamic environments and structural dynamics of helicopters through a comprehensive flight test program of an extensively instrumented AH-1G. A comprehensive data base was acquired of rotor aerodynamic forces, aeroelastic loads, blade motions, acoustics, and the attendant responses of the control system and airframe that result from flying operational manuevers. Continuous and simultaneous data were recorded from 387 transducers for 224 different flight conditions. Flight test conditions were chosen to obtain data over the maximum range of rotor loadings possible within the cost and time frame restrictions. Over 72,000 separate variable time histories were digitized and recorded on magnetic tape and made available through an interactive computer program at the user's command.

Strain gages were used to measure blade bending moments and strain in mechanical components such as the rotor shaft, pitch links, and blade grip. The blade was strain gaged at ten radial stations providing complete distributions of beam bending, chord bending, and torsion along the blade radius; 110 rotating pressure transducers along the radius provide forces and moments experienced by the rotor. In addition, 35 accelerometers were installed at various locations on the fuselage, rotor blades, and pylon. This data base thus provides the unique ability to see complete maps of rotor loads and simultaneous loads at other key locations on the helicopter airframe and control system while the helicopter was performing operational manuevers. The loads in the various components of the aircraft can be related to the manuevers, which may lead to a better understanding of load spectrum definition and application.

A portion of this flight test was devoted to flying a spectrum derived for NOE operations. The purpose of this NOE data acquisition and analysis was to determine if the published recommended fatigue lives were reduced by NOE operations. The NOE loads were compared to the existing TOW missile and in ground effect (IGE) manuever loads obtained from the AH-1F/S qualification loads surveys. All of the measured dynamic parameters were reviewed and the NOE loads compared favorably with the published values. As a result, it can be concluded that the NOE operation, as outlined in this program, does not reduce the published retirement lives on the dynamic components of the AH-1G helicopter.

The OLS data base has also provided the opportunity to conduct detailed correlation of the experimental test data with various helicopter computer simulations. The accuracy and limitations of the analytical models have been and are continuing to be evaluated in order to provide the analyst with confidence in his ability to predict loading conditions. Illustrations of these correlations are provided in Figures 10 and 11.

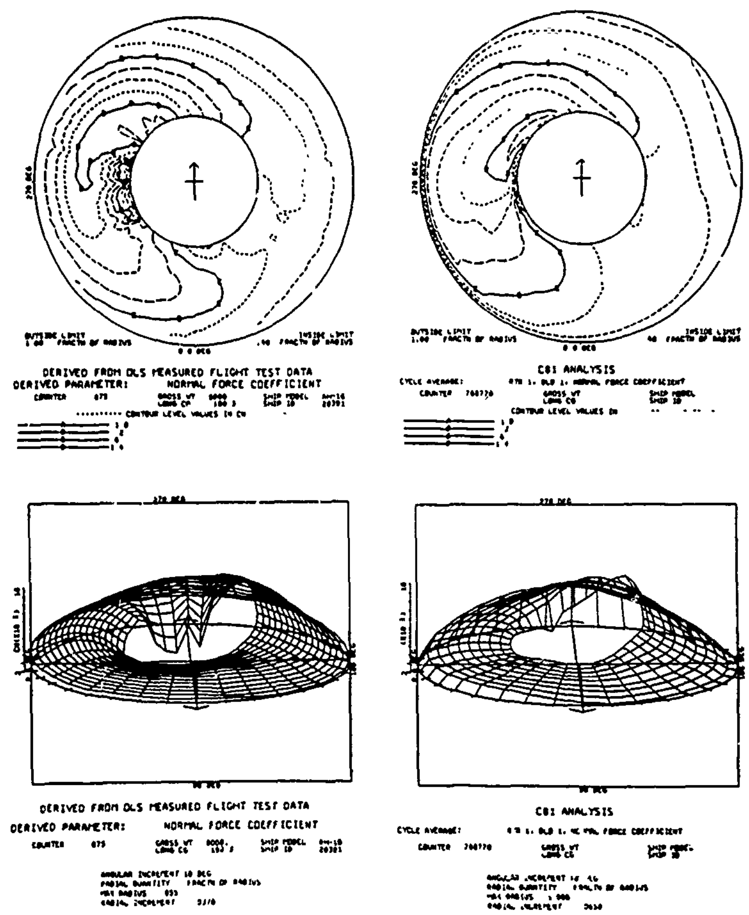

Figure 10. Contour and Surface Plots of Computed and Derived Main Rotor Normal Force Coefficients.

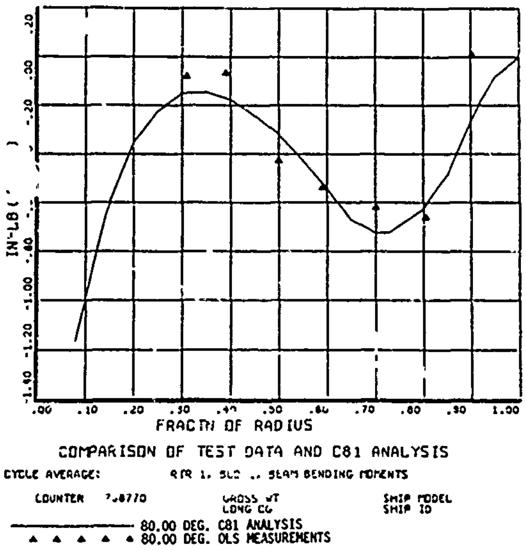

Figure 11. Radial Comparison of Measured and Computer Blade Beam Bending Moments at 80 Degrees Azimuth

MISSION PROFILES

The Reference 25 study developed a standard mission profile for all classes of helicopters. The profile is comprised of six mission segments, with each segment containing basic (and ground) conditions under which the helicopter would be expected to operate. Each basic condition may encompass several detailed flight conditions. For example, the turn condition which appears in segments 2 through 6 may encompass right, left, S, and steep turns with varying rates of entry and recovery. A main distinction of the segments, compared to those used in the measurement programs, is that they do not include a separate maneuver segment. This was done in recognition of the fact that maneuvers occur in all phases of flight.

Mission profiles were developed for six helicopter types, based upon measured data from operational usage programs. The profiles for four aircraft types are presented in Table 2 and compared to profiles from CAM-6, AMCP 706-203, and AR-56. The profiles presented from the specifications were not to the same format as in the table. Therefore, the basic conditions of the three specification profiles were broken down into conditions compatible with the developed profiles.

Ground Operations: The percentage values assigned to ground operations in the developed profiles are greater than those from AR-56 and CAM-6. The specification profiles consider ground conditions very generally, and it was difficult to establish precisely what was classified as a ground condition. When the percentage values assigned to ground conditions were adapted to the standard profile format, the time percentages for specific conditions seemed low, especially when connected to units of flight time.

Takeoff, Landing, and Low-Speed Flight: The developed profiles compare reasonably well with CAM-6 and AMCP 706-203. However, values from AR-56 are considerably higher, particularly for the utility- and attack-type helicopters. The major contributor to the differences appears to be the values assigned to steady hover. In many helicopter operations steady hover is a momentary flight condition occurring prior to the transition from takeoff to forward flight. The exceptions are helicopters involved in load lifting operations (crane) and stationary weapons delivery (attack). In the case of the attack helicopter, the derived profile was predominantly from SEA operations in which the operational tactics were unique to that theater and did not include mission assignments, such as an antitank mission which would increase hover time to be more commensurate with specification percentages.

Ascent Conditions: The developed profiles contained more time allotted to this condition than specification profiles. Operational reports supported this.

Forward Flight Conditions: The specification and developed profiles overall compare closely for this segment. The developed profiles do show less time in steady level flight, transient conditions of acceleration and deceleration, and yawed flight and more time in turns than the specification profiles.

Descent Conditions: The overall percentage times are comparable between the referenced and developed profiles.

Autorotation: With the exception of CAM-6, the percentage of occurrence values for the autorotation mission segment for all the profiles compared quite closely.

Distribution of load factor with airspeed, gross weight, and altitude was also constructed for each developed profile.

TABLE 2. Mission Profile Comparison

Condition	CAM-6	AMCP 706-203	Utility AR-56	Utility Derived	Attack AR-56	Attack Derived	Crane AR-56	Crane Derived	Transport AR-56	Transport Derived
1 GROUND OPERATIONS										
A. Startup	0.10	2.59	0.10	0.50	0.10	0.50	0.10	0.50	0.10	0.50
B. Shutdown	0.10	1.29	0.10	0.50	0.10	0.50	0.10	0.50	0.10	0.50
C. Ground run	0.28	2.92	0.20	2.00	0.20	2.00	0.20	2.00	0.20	2.00
D. Taxi	0.48	1.00	0.48	1.00	0.48	1.00	0.48	1.00	0.48	1.[illegible]
	0.96	7.80	0.88	4.00	0.88	4.00	0.88	4.00	0.88	4.00
2 TAKEOFF/LANDING/LOW-SPEED FLIGHT (<40 knots)										
A. Vertical liftoff	0.88	0.40	0.58	1.58	0.58	0.22	0.58	0.51	0.58	0.66
B. Rolling takeoff	0.40	0.26	0.40	0	0.40	0.27	0.40	1.24	0.40	1.53
C. Vertical landing	1.35	0.59	0.80	1.58	0.80	0.22	0.80	0.51	[illegible]	0.66
D. Slide-on landing	1.34	0.11	0.20	0	0.20	0.27	0.20	1.24	0.70	1.53
E. Hover (steady)	0.45	3.45	8.80	2.81	4.48	0.69	14.08	6.18	10.56	3.52
F. Hover control reversals	1.24	0.07	0.73	0.31	0.29	0.25	0.73	[illegible]	0.73	0.63
G. Hover turns	0.20	0.13	1.47	0.31	0.59	0.25	1.47	1.49	1.47	1.22
H. Pop-ups	0.03	-	-	0	0.02	0.48	-	[illegible]	-	0
I. Sideward flight	0.48	0.10	0.88	0.31	0.88	0.25	0.88	[illegible]	0.87	0.43
J. Rearward flight	0.48	0.03	0.44	0.16	0.44	0.13	0.44	0.61	0.44	0.23
K. Low-speed forward flight (air taxi)	1.70	5.00	4.40	2.78	2.20	2.77	3.52	1.12	4.40	4.75
L. Flare	1.43	0.52	0.80	1.59	0.80	0.48	0.80	0.53	0.80	2.45
M. Vertical climb	0.50	0.40	1.32	0.42	1.32	0.69	1.76	2.24	1.32	0.29
N. Vertical descent	0.50	0.50	0.80	0.42	0.80	0.69	0.80	2.24	0.80	0.29
O. Low-speed turns	0.70	0.43	1.10	0.74	2.[illegible]	0.87	1.10	0.61	1.10	1.22
	11.68	11.90	22.72	12.92	15.80	8.53	27.56	20.89	24.48	19.21
3 ASCENT (>40 knots)										
A. Steady-state climb	5.20	4.00	2.20	9.00	2.20	3.66	4.40	9.91	2.20	9.83
B. Turns	2.00	0.40	1.10	2.32	2.00	4.34	1.10	0.65	1.10	0.99
C. Pushovers	0.20	0.05	-	0.3[illegible]	-	1.16	-	0	-	0
	7.40	4.45	3.30	11.66	4.20	9.16	5.50	10.56	3.30	10.82
4. FORWARD FLIGHT (>40 knots)										
A. Level flight (40, 60, 80, 100% V_H)	51.40	62.81	56.85	39.02	57.78	33.40	50.17	47.58	55.19	46.28
B. Turns	3.00	1.38	1.10	9.91	2.05	11.65	1.10	3.02	1.10	3.43
C. Control reversals	1.44	0.06	0.59	1.27	1.47	0.63	0.59	0.10	0.59	1.51
D. Pull-ups	0.95	0.30	0.18	0.25	0.37	2.85	0.18	0.05	0.18	0.10
E. Pushovers	0.75	0.12	-	0.03	-	2.85	-	0	-	0.01
F. Deceleration	2.83	1.50	0.46	2.55	0.46	2.85	0.46	0.30	0.46	0
G. Acceleration	4.59	1.62	1.10	2.55	0.44	2.85	1.10	0.30	1.10	0
H. Yawed flight	1.56	0.55	3.00	0.34	3.04	0.51	2.64	0.08	2.90	0.10
	66.52	68.34	63.28	55.92	65.61	57.59	56.24	51.43	61.52	51.43
5. DESCENT (power on, >40 knots)										
A. Partial power descent (steady descent)	1.40	4.34	4.89	5.26	4.90	2.21	4.89	9.10	4.89	7.97
B. Dive (power on, 115 percent V_H)	3.00	1.21	1.98	3.50	2.64	3.45	1.98	0.05	1.98	2.12
C. Turns	1.50	0.73	1.10	3.31	2.00	9.27	1.10	0.61	1.10	1.09
D. Pull-ups	0.97	0.59	0.03	0.10	0.64	2.46	0.03	0.03	0.03	0.03
	6.85	6.87	8.00	12.17	10.18	17.39	8.00	9.79	8.00	11.21
6 AUTOROTATION (power off)										
A. Entries (including power chops)	1.43	0.02	0.03	0.39	0.07	0.39	0.03	0.39	0.03	0.3[illegible]
B. Steady descent	1.90	0.05	0.88	1.63	1.76	1.63	0.88	1.63	0.88	1.63
C. Turns	0.40	0.04	0.36	0.55	0.88	0.5[illegible]	0.36	0.55	[illegible]36	0.55
D. Power recovery	0.48	0.5[illegible]	0.03	0.24	0.07	0.24	0.03	0.24	[illegible]	0.24
E. Flare and landing	2.38	0.01	0.26	0.52	0.26	0.52	0.26	0.52	[illegible]	0.52
	6.59	0.64	1.56	3.33	3.04	3.33	1.56	3.33	1[illegible]	3.33

PEAK PARAMETER COMPARISONS

Table 3 presents peak parameter values recorded in select operational measurement programs for attack, observation, crane, and transport helicopter types. For comparative purposes, peak values are also presented from the structural design criteria for each helicopter. Except for the cargo helicopter, all measurement programs are exclusively from SEA operations; the cargo helicopter included both USA and SEA operations. Except for a few specific peak loads, only approximate peak values could be derived, since most of the recorded points represented ranges of the measured parameter, making it impossible to establish the exact peak value within the range.

TABLE 3. Peak Value Comparison - Design vs. Operations Data

PARAMETER	ATTACK		OBSERVATION		CRANE		TRANSPORT	
	DESIGN	OPERATIONAL	DESIGN	OPERATIONAL	DESIGN	OPERATIONAL	DESIGN	OPERATIONAL
AIRSPEED, KNOTS	222	185	130 V_{NE}	124-130	126.5 V_d	132	–	–
VERTICAL ACCEL, g								
POSITIVE	2.43	2.40	2.55	2.20-2.40	2.26	1.50, 1.88 (Hoist)	2.70	1.70
NEGATIVE	-0.50	0.10	-0.50	0.0-0.20	-0.50	0.50	-0.50	0.20
ENGINE TORQUE PRESSURE, PSI	62.5	60-70	80.3	80-90		–	–	–
RPM								
MAXIMUM	356	351	514	490	204	203	–	–
MINIMUM	294		470	440	–	–	–	–
GROSS WEIGHT, LB	9,500	9,522	2,400	>2,600	42,000	44,009	–	–

Of the maximum one-time values measured for the AH-1G attack helicopter, the gross weight exceeded the value specified in the structural design criteria report, but by an insignificant amount. Pilots and observers in SEA report that the AH-1Gs frequently took off with the maximum liftable weight. Because of the higher temperatures in this operational theater and the associated adverse effect on helicopter power, there was little opportunity to seriously overload the aircraft. Such may not be the case, however, in a cooler environment. The maximum value for engine torque recorded was greater than 60 and less than 70 psi, thus possibly exceeding the 62.5 psi structural design criteria value. Based on other recorded parameters during this occurrence, the most probable event was the requirement to take off over an obstacle under conditions requiring high power.

The OH-6A observation helicopter measured peak parameter values exceeded the design criteria maximums in gross weight and possibly engine torque.

For the CH-54A crane helicopter, the maximum recorded airspeed was approximately 120 knots indicated. For a density altitude of 5,000 feet at which this airspeed was measured, the equivalent true airspeed would be 132 knots, which exceeds the limit dive speed of 126.5 knots. The maximum recorded gross weight during SEA operations was 44,009 pounds which exceeded the basic design gross weight by over 2,000 pounds. Field studies conducted in SEA indicated that, during certain operations, more than 30 percent of the sorties were flown at gross weights in excess of the design values. Overload gross weights were rarely flown at speeds higher than 60 to 70 knots or at load factors greater than 1.0 ±.2g. This practice of flying at overload gross weights, although possible on the basis of static strength, may have produced flapping, power, or control requirements approaching fatigue damaging levels.

The only peak value comparison made for the CH-47A transport helicopter was vertical load factor wherein the peak measured value was considerably below the design limit load value. Reference 19 cautions against concluding that a lower design load factor value is appropriate because of the uncertainty of adequacy of the data sample to represent long time exposure of a full fleet of aircraft. The maximum (or minimum) values recorded during a relatively small usage monitoring period involved in data acquisition are related to the probability of occurrence of greater values over a period more representative of a fleet's service life. References 17, 19, and 20 suggest extrapolation of load factor exceedance curves, as well as exceedance curves for other parameters, as a practical approach for extending the load factor data key and sample time. With this extrapolation, design values will be more closely approached and in some instances possibly exceeded. An obvious issue is how far to extrapolate to

represent fleet usage, yet at the same time keep the extrapolation valid. Reference 19 indicates that the design load factor for the CH-47A was based on the thrust capability of the rotors. The demonstrated load factors on the CH-47 models were the extremes that the pilot could attain for the required demonstrations. Therefore, if the required demonstrations were properly chosen and demonstrated, it should not be possible for a fleet pilot to exceed the demonstration values. This reasoning leads to the conclusion that in the case of load factor, the extrapolated value should become asymptotic to the demonstrated value.

MONITORING SYSTEMS OF FATIGUE DAMAGE TO HELICOPTER COMPONENTS

As data from operational usage measurement programs became available, it became desirable to compare them to the usage spectra to which the helicopter was originally designed and to ultimately assess the import of any differences on the fatigue lives of flight-critical structural components. To this end, several studies were conducted. In References 1 through 5 the four mission segment breakdowns from the usage measurement programs were restructured into specific flight regimes such as ground, hover, takeoff maneuvers, sideward and rearward flight, and gust. In that the four segment breakdown did not provide enough information to derive specific conditions. The contractor had to make judgmental decisions based on existing profiles and company experience with the aircraft as well as others in its class.

Recognizing this difficulty in using the existing measured data from service usage programs for detailed fatigue analysis of structural components, the US Army had the Reference 21 study conducted in 1973 to investigate methods for monitoring and recording the in-flight operation of helicopters for specifically assessing fatigue damage to their structures. Four concepts for in-flight aircraft monitoring were considered: flight condition monitoring (FCM), component load monitoring (CLM), direct monitoring (DM), and mission type monitoring (MTM).

FCM involves recording flight parameters to identify flight conditions such as hover, forward level flight, and turns; these conditions form the basis for fatigue damage assessment. Similar to most manufacturers' fatigue analysis, where damage is assigned to various flight conditions of a design spectrum, the fatigue analysis with FCM is based on the actual flight time spent in various flight conditions. The fatigue damage assessment is based upon a damage rate per unit of time for each selected flight condition; that is,

$$D(I) = C(I) \times T(I)$$

where
D = damage for a given flight condition
C = damage rate for a given flight condition
T = total time spent in a given flight condition

Therefore, in comparison to the classical method of computing fatigue damage where applied cycles (n) of specific load levels are compared with the corresponding number of cycles to failure (N), the FCM concept is based on the assumption that the actual time spent in a flight condition can be used to identify the applied cycles and that the number of cycles to failure can be established for the flight condition.

CLM involves recording component loads either directly or indirectly and using these loads to assess fatigue damage. This method of assessing fatigue damage is comparable to the manufacturer's analysis where component loads are related to various flight conditions. In the CLM concept, however, the recorded loads need not be associated with flight conditions since the loads are recorded on a real-time basis. Like the FCM, the CLM records elapsed time within a specified range of the component load. Consequently, the fatigue damage is again assessed on a rate basis:

$$D(I) = C(I) \times T(I)$$

where
D = damage for a given load range
C = damage rate for the given load range
T = elapsed time spent in the load range

This damage assessment is similar to the classical method of computing fatigue damage. The applied cycles (n) of the loads in the range are derived from T(I) and the damage rate (1/N) is conservatively based on the maximum load in the load range. Ideally, the monitored loads would be the critical loads of the fatigue-critical components. This, however, results in costly and complex instrumentation requirements: slip rings, multiplexing equipment, and circuitry. A compromise to these requirements is expressing the loads on the dynamic components on the helicopter rotor system as a linear function of the loads in the stationary control system.

The DM method uses data gathered on various fatigue-related phenomena to assess fatigue damage. The phenomena include the change in metal conductivity due to cold working, the acoustical emission of materials under stress, and the change in magnetic field strength as a result of fatigue loads. Unlike the FCM and CLM methods, DM empirically associates the monitored data with the accrued fatigue damage. This association would be determined through tests of full-scale components measuring both the monitored data and the accrued component fatigue damage. From these tests, the criteria for retirement lift projection and component removal would be established in terms of the monitored data.

MTM involves tracking each helicopter's functional mission assignment which is used to assess the fatigue damage to the critical structural components. These mission assignments are part of the Army's maintenance reporting system and typically include direct combat training, direct combat support, aeromedical, flight training, and technical operations and maintenance training. The fatigue damage of the structural component is assessed according to the flight time spent by the aircraft in each functional assignment. The damage would be assessed similarly as in the FCM method, i.e., computed as the product of flight time and a theoretical damage rate for the functional assignment under consideration. Damage assessment with this method requires that a theoretical damage rate be determined for each functional assignment. Determination of the damage rate requires such information as the definition of typical mission profiles, the frequency of occurrence of flight conditions for the mission profiles, and the corresponding component flight loads for each functional assignment. Since this information was not available, an alternative MTM method based on a mission segment concept was also investigated. This alternate concept assumes that the mission profile can be subdivided into several segments (ascent, descent, etc) and that fatigue damage to the components can be assessed for the flight time spent in the various mission segments. As in the FCM method, the theoretical damage rates can be determined by considering the flight conditions that the mission segments would include. Each of the MTM methods has its advantages. The functional assignment technique is an extremely simple monitoring method that could easily be incorporated into the Army's maintenance reporting system. The disadvantage is the difficulty in identifying of the theoretical damage rates for the functional assignments. The alternate method would be more complex, requiring an onboard recorder, but would provide better information for fatigue damage assessment.

Each of the four monitoring concepts was evaluated for technical acceptability and cost effectiveness for a UH-1H utility and a CH-47C cargo helicopter. Results indicated that FCM is a superior overall method for each aircraft type. This was only true for the CH-47C provided that the monitoring of gross weight becomes technically acceptable. In the past, reliable and accurate measurements of helicopter gross weight had not been shown to be feasible in an operational environment. Although not selected, the CLM and DM methods would potentially monitor fatigue damage better than the FCM or MTM methods since the data recorded by the former two methods would be closely associated with the fatigue phenomenon. For the CLM, the development of transfer functions relating rotating component loads to stationary component loads should be investigated. A flight test program, where the transfer functions could be derived empirically, could be used.

In the process of providing flight condition data for fatigue analysis, the FCM method defines how the helicopter is being used, thus providing insight into the types of operations resulting in the associated fatigue damage rates. Such utilization data establishes a useful base for model changes and new helicopter system developments. This method also provides a more versatile operational data set for uses other than fatigue such as maintenance analysis. Recently, there has been consideration of a multifunctional recorder system to acquire operational data for fatigue analysis, maintenance actions, and even accident information. Although the data editing and analysis for each function would vary, the measured flight parameters would be common, to a large degree, for all three functions. Thus, an FCM concept for fatigue analysis would be the best approach for such a system.

As the afore mentioned monitoring system study concluded, the CLM method is a more direct concept for fatigue analysis since it eliminates an intermediate step of load definition for particular flight conditions; it also eliminates the need for gross weight measurement. The CLM approach was investigated in a Canadian study for the Boeing Vertol CV 113 and is currently under evaluation in a US program. The major drawbacks of this approach are the necessity for strain gages, which tend to be easily damaged and unreliable in service applications, and for slip rings if measurements must be made in the rotating system. These drawbacks could be alleviated by developing rotating to fixed system transfer functions to eliminate slip ring requirements, using alternate rotating to fixed signal transfer methods, and keeping the number of gages to a minimum.

Perhaps the answer to recording methods lies in an approach which is a hybrid of the FCM and CLM concepts. Parametric measurements can be made to define operational usage of interest and direct load measurements can be taken on key structural components. With emerging onboard microprocessor recorder technology, this should be technically feasible--the constraint may, however, be recorder cost.

STRUCTURAL INTEGRITY RECORDING SYSTEM

Based upon the findings and recommendations from the fatigue monitoring feasibility study, the Army in 1975 began developing a method for tracking the accumulation of fatigue damage on critical helicopter dynamic components. The Structural Integrity Recording System (SIRS), as the system is called, relies on a flight condition recognition (FCR) method for monitoring the variations in fleet utilization on a helicopter-by-helicopter basis. SIRS is a total system comprising an airborne microprocessor based recorder, a portable flight line retrieval unit, and a data processing package (see Figure 12). The recorder monitors various flight parameters and stores preselected types of operational data within the recorder's solid-state memory. Data are retrieved by a portable flight line retrieval unit that transfers the recorded data onto removable, miniature computer compatible tape cassettes. Each cassette can store the average monthly operational data of approximately 50 helicopters. At a central data

processing site, the software system automatically processes and analyzes the data, and then generates tailored reports that present the usage and corresponding incremental fatigue damage to each component for each monitored helicopter. The SIRS was developed initially for the AH-1G helicopter.

The FCR method of fatigue damage assessment is structured as follows: Defined in terms of specific combinations of flight parameter ranges, each flight condition category (FCC) represents one or more flight conditions. The component damage due to each flight condition may be determined when the loads during the flight condition, the number of flight occurrences, and the component fatigue strength are known. To ensure that the damage rate for each flight condition category is conservative, the maximum flight condition damage rate within the given flight condition category is chosen. Then the component damage accrued during a given recording period may be computed by Equation (1), and the flight condition category incremental damage may be summed to yield the total component damage. The total recorded time is calculated by Equation (2), and the fatigue life is predicted by Equation (3).

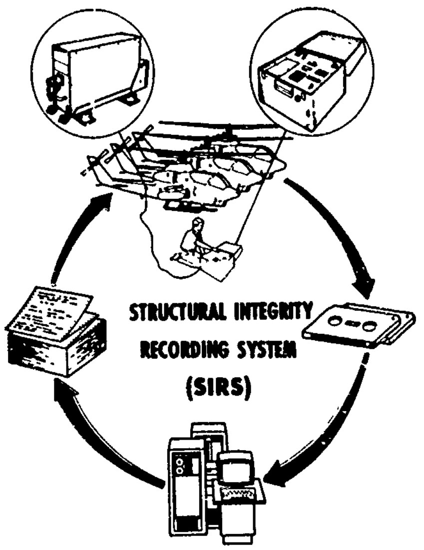

Figure 12. SIRS Data Retrieval and Analysis

$$D = \sum_{k=1}^{m} D_k = \sum_{k=1}^{m} C_k T_k \qquad (1)$$

$$T_t = \sum_{k=1}^{m} T_k \qquad (2)$$

$$FL = \frac{T_t}{D} \qquad (3)$$

where D = total damage to a component during the usage spectrum

D_k = component damage accrued during the kth flight condition category

C_k = damage rate in kth flight condition category for a particular component

T_k = amount of flight time spent in kth flight condition category

T_t = total flight time

FL = component fatigue life

m = number of flight condition categories

The FCM method of fatigue damage assessment requires analyzing the manufacturer's fatigue analysis for the helicopter for which the system is to be used to first define a technically feasible FCM system and then to establish damage rates for each component in each flight condition category. After these data have been developed and substantiated, the selected flight parameters may be monitored to assess the accrued fatigue damage of critical helicopter dynamic components.

TABLE 4. SIRS Parameters

- **MEASURED**
  - PITOT PRESSURE
  - STATIC PRESSURE
  - OUTSIDE AIR TEMPERATURE
  - TRANSMISSION LIFT LINK STRAIN
  - AIRCRAFT PITCH ANGLE
  - AIRCRAFT ROLL ANGLE
  - ENGINE TORQUE PRESSURE
  - MAIN ROTOR RPM
  - VERTICAL ACCELERATION AT CG
- **COMPUTED**
  - PERCENT V_L AIRSPEED
  - PERCENT V_H AIRSPEED
  - DENSITY ALTITUDE

The airborne recorder shown in Figure 12 is made possible by the application of a microprocessor to provide a programmable system. The microprocessor monitors the parameters in Table 4. When these flight parameters fall in preset ranges or form certain flight conditions, the microprocessor accumulates their occurrence or the amount of time associated with them in the recorder data storage memory. The flight conditions are defined generally as various combinations and sequences of the measured parameters. The SIRS recorder processes the inputs from the transducers for the monitored parameters. Each input is conditioned to a desired full-scale signal level, multiplexed, and converted from an analog to a digital signal that can be manipulated by the micorprocessor under program control. The recorded data is in three forms: (1) time in various flight conditions, (2) number of occurrences of other flight conditions, and (3) maximum value of some input parameters. The recorder incorporates a Lithium Organic Battery with a minimum of

1-year's capacity to retain the stored data when aircraft power is off. The complete recording system installation weight is 20.6 pounds. The recorder, including monitoring rack and electromagnetic interference shielded connector, is 17.5 inches long, 6.5 inches wide, and 10.15 inches high.

SIRS is designed so that data need be retrieved only once every 4 to 6 weeks by a portable, flight-line data retrieval unit (RU). During transcription of recorded data onto the miniature tape cassette, the operator interacts with the unit. The RU displays messages and the operator communicates with the unit through a keyboard. The data retrieval, including setup, takes less than 5 minutes and can be performed on a flexible schedule. In addition to data retrieval, the RU performs diagnostic checks of the recorder and transducers. It can also be used as a readout device during transducer calibrations.

The data retrieval unit is 19.1 inches long, 15.6 inches wide, and 9.8 inches high. Containing its own rechargeable power system, the retrieval unit is housed in a flight line style container.

Upon receiving the data from the miniature cassettes, the software system first performs an initial data processing to verify the recorder operation and transducer functioning and to review the long-term trend of the transducer static readiness, and then analyzes the data. The analysis includes segregation of the data by flight condition categories, conversion to a 100-flight-hour basis, and data presentation in terms of usage spectrum. Next, the software system analyzes the data by calculating the incremental fatigue damage for specific components.

A SIRS recorder and transducer package was installed on an AH-1G at the Army Aviation Test Board, Fort Rucker, Alabama. In addition, a parallel oscillograph recording system was installed. The aircraft was then flown through all the maneuvers in the usage spectrum. Data was processed from the two systems and compared. Good correlation was obtained for most of the flight conditions. Some requirements for recorder software changes were identified. These changes were made before the next phase of testing. The SIRS was installed on five aircraft at the Army-Aviation Center, Fort Rucker, Alabama. Active data collection over a 3-month period with a total of 260 hours of flight was monitored by SIRS.

In October 1977, a follow on effort was initiated to install five SIRS recorders on AH-1S aircraft. Software modifications were made to adapt the system from the G to the S COBRA model. Hardware modifications were made to increase the memory fourfold. This was done to permit recording more data. A tri-variant table of lateral control position, engine torque, and airspeed was attached to the transmission lift links to measure of aircraft gross weight in flight. This was in lieu of a strain-gaged lift link used on the AH-1G which had been found to be unreliable under field conditions.

Several problems have been identified during the AH-1S flight test program that required changes to the hardware and software to provide a satisfactory system. The major problem is the gross weight sensing system which is also used to detect lift-off. The piezo-electric strain sensor is extremely sensitive, as required to measure the lift link strain. However over a period of time in field usage, drift occurs which makes it output unusable by the recorder. A method of determining lift-off from a combination of roll position and engine torque has been developed but not yet verified.

Table 5 presents the damages calculated for dynamic components AH-1S aircraft during a 6-month period. The table shows individual and total damage fractions for seven components. The delta logbook hours, delta recorder flight time hours, and delta recorder ground time hours were added for clarity. The three damage fraction columns are labeled "SIRS", "RECORDER", and "LOG". The damage under the "SIRS" column results from the calculations performed using the recorded flight condition category items and the associated damage rate coefficients. The damages under the "RECORDER" column represents the assumed damage based upon the recorded flight time multiplied by the inverse of the recommended retirement life of the component (i.e., the fractional portion of the retirement life that has been used based upon recorded flight time). The damage under the "LOG" column represents the assumed damage or the fractional portion of the retirement life used based upon the logbook hours.

TABLE 5. Calculated Component Damage

COMPONENT	SIRS	RECORDER HOURS	LOG BOOK HOURS
MAIN ROTOR BLADE	.07264	.07392	.13009
MAIN ROTOR YOKE EXTENSION	.00092	.02465	.04335
MAIN ROTOR GRIP	.00000	.00813	.01431
MAIN ROTOR PITCH HORN	.00062	.01231	.02168
RETENTION STRAP NUT	.09286	.03696	.06504
SWASHPLATE DRIVE LINK	.00023	.00739	.01301
SWASHPLATE OUTER RING	.00080	.02465	.04335
TAIL ROTOR BLADE	.00000	.00000	.00000

CONCLUSIONS AND RECOMMENDATIONS

The mission assignment of a helicopter is probably the most influential factor in establishing the character of a flight spectrum. The variability in flight spectra obtained for operational helicopters flying several different mission assignments clearly demonstrates this point.

Fatigue mission profiles for structural component life calculations should be based on design flight conditions and reevaluated by conducting surveys of actual helicopter operations.

SIRS has demonstrated the effective use of an onboard microprocessor recorder for operational fatigue data measurement.

Results from the service usage programs and studies directed at using these results to update design criteria and fatigue analysis have identified the following further considerations for service usage monitoring the structural integrity recording.

- Composite mission profile based upon a complete cross section of operating conditions.

  Additional assessments are recommended to examine the need for a weighted design spectrum that would encompass the full use of a helicopter in US Army operations. It would add to overall accuracy and realism in helicopter design. The data sample for creating this spectrum should include a complete cross section of operating conditions for different operational units in various parts of the world. An operational mission profile, developed solely from one data set such as SEA, does not provide the absolute design tool. Certain special and limiting conditions which were prevalent during SEA operations affected the results. Such conditions may include level of command from which the helicopters are deployed, mission assignment, aircraft availability, and flight tactics.

- Well-defined discrete ground and flight regimes.

  The particular assessment needs for which the data will be used should be well defined and the measurement list and parameters tailored to meet the needs. Several reports have recommended smaller increments and inclusion of data closer to the 1g level and the presentation of load factor data to allow determination of time at equal load factor increments.

- Meaningful combinations of recorded data.

  Individual specific load parameters, such as rate of climb, cross-plotted against time for each important load parameter such as load factor, airspeed, and gross weight would be valuable in better defining flight regimes and detailed flight conditions.

- Direct monitoring of loads in fatigue critical components.

  The availability of operational load parameters for specific components would be beneficial in establishing operational trends. Valuable information would be gained for future main and tail rotor fatigue designs if data were available for main rotor blade flapping, tail rotor flapping, and collective.

- Maximum one-time-occurrence data.

  When evaluating the cause for the limit of a particular parameter, it would be desirable to have the instantaneous value of all other measured parameters, or even better, a short time history of the event. This would aid in reconstructing the condition and thus give credibility to the argument for the cause.

Acquisition and presentation of helicopter operational data has improved greatly since the early efforts in the 1950s. In spite of this improvement, a continuing concerted effort is required to characterize the usage of both mature and new helicopters. The latter have greater performance and structural capabilities, and new intended usages. The need for continued monitoring, coupled with ever-improving solid-state microprocessor recorder technology, lead to the future prospect for meaningful, worthwhile usage monitoring efforts.

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Appendix - US Army Helicopter Service Usage Programs

	UH-1B	UH-1H	UH-1H	UH-1H	UH-1H	AH-1G	AH-1G
PROGRAM	UH-1B Helicopter Flight Loads Investigation	Operational Use of UH-1H in Southeast Asia	Operational Use of UH-1H in Artic Enviroment	Structural Loads Survey During Cold-Weather Operations	Acquisition of Operational Data During NOE Missions	Operational Use of AH-1G in Southeast Asia	Operational Loads Survey
REPORT REFERENCE	USAAVLABS TR-66-46	USAAMRDL TR-73-15	USAAMRDL TR-74-65	USAAMRDL TR-75-3	USRTL TR-78-3	USAAMRDL TR-70-51	USAAMRDL TR-76-39
LOCATION	Fort Benning, Georgia, USA	Southeast Asia (Vietnam)	Fort Greely, Alaska, USA	Fort Greely, Alaska, USA	Fort Rucker, Alabama, USA	Southeast Asia (Vietnam)	Arlington, Texas, USA
MISSION	Combat Training	Combat	Artic	Artic	Nap-of-the-Earth	Combat	Test
FLIGHT HOURS	219	203	88	18	14.9	408	28
INSTRUMENTED AIRCRAFT	4	3	2	1	1	5	1
TIME FRAME	Aug 64 - May 65	Sept 71 - Mar 72	Dec 72 - Apr 73	Jan 74 - Feb 74	Dec 76 - Apr 77	Jul 68 - Jan 70	Jun 75 - Sep 75
INSTRUMENTATION:	Oscillograph Recorder	Oscillograph Recorder	Oscillograph Recorder	Oscillograph Recorder	Oscillograph Recorder	Oscillograph Recorder	A total of 387 Channels on Magnetic tape
Outside Air Temp	X	X	X	X	X	X	X
Pressure Altitude	X	X	X	X	X	X	X
Engine Torque	X	X	X	X	X	X	X
Forward Airspeed	X	X	X	X	X	X	X
Lateral Airspeed					X		X
Collective Position	X		X	X	X	X	X
Long. Cyclic Position	X		X	X	X	X	X
Lat. Cyclic Position			X	X	X		X
Rudder Pedal Position			X	X	X		X
Yaw Attitude					X		X
Yaw Rate					X		X
Pitch Attitude					X		X
Pitch Rate					X		X
Roll Attitude					X		X
Roll Rate					X		X
Nx at CG	X	X	X	X	X	X	X
Ny at CG	X	X	X	X	X	X	X
Nz at CG	X	X	X	X	X	X	X
Ny at Tail					X		X
Main Rotor Speed	X	X	X	X	X	X	X
Exhaust Gas Temp					X		
Heading					X		X
MR Chord Bending				X			X
Mr Beam Bending				X			X
TR Chord Bending				X			X
TR Beam Bending				X			X
Drag Brace Load				X			X
Scissors Link Load				X			X
Long. Boost Tube Load				X			X
Lat. Boost Tube Load				X			X
Coll. Boost Tube Load				X			X
MR Azimuth				X			X
TR Azimuth				X			X
Time	X	X	X	X	X	X	X
Indicated Airspeed					X		X
Sonic Altitude					X		
TR Shaft Torque				X			X

	CH-47A	CH-47A	CH-47A	CH-54A	CH-54A	OH-6A
PROGRAM	CH-47A Chinook Loads Investigation Program	Flight Loads Investigation of cargo & Transport CH-47A Helicopters Operating in Southeast Asia	Flight Loads Investigation of Combat Armed & Armored CH-47 Operating in Southeast Asia	CH-54A Skycrane Helicopter Flight Loads Investigation Program	Flight Loads Investigation of CH-54A Helicopters Operating in Southeast Asia	Flight Loads Investigation of OH-6A Helicopters Operating in Southeast Asia
REPORT REFERENCE	USAAVLABS TR-66-68	USAAVLABS TR-68-2	USAAVLABS TR-68-1	USAAVLABS TR-68-58	USAAVLABS TR-70-73	USAAMRDL TR-71-60
LOCATION	Fort Benning, Georgia, USA	Southeast Asia	Southeast Asia	Fort Benning, Georgia, USA	Southeast Asia	Southeast Asia
MISSION	Cargo & Transport	Cargo & Transport	Aerial Fire Support & Reconnaissance	Routine Cargo	Cargo Transport	Combat
FLIGHT HOURS	165	235	207	110.6	410	216
INSTRUMENTED AIRCRAFT	6	4	3	3	3	3
TIME FRAME	Sept 64 - Dec 65	Jan 66 - May 67	Aug 66 - May 67	Feb 65 - Jul 65	Aug 68 - Feb 70	Mar 70 - Sep 70
INSTRUMENTATION:	Oscillograph Recorder	Oscillograph Recorder	Oscillograph Recorder	Oscillograph Recorder	Oscillograph Recorder	Oscillograph Recorder
Outside Air Temp	X	X	X	X	X	X
Pressure Altitude	X	X	X	X	X	X
Engine Torque		X		X	X	X
Forward Airspeed	X	X	X	X	X	X
Lateral Airspeed						
Collective Position	X	X	X	X	X	X
Long. Cyclic Position	X	X	X	X	X	X
Lat. Cyclic Position						
Rudder Pedal Position						
Yaw Attitude						
Yaw Rate						
Pitch Attitude						
Pitch Rate						
Roll Attitude						
Roll Rate						
Nx at CG			X		X	X
Ny at CG			X		X	X
Nz at CG	X	X	X	X	X	X
Ny at Tail						
Main Rotor Speed	X	X	X	X	X	X
Exhaust Gas Temp		X				
Time	X	X	X	X	X	X
Gas Producer RPM		X		X		

HELICOPTER DATA ACQUISITION IN WHL

P S GRAINGER - CHIEF MECHANICAL ENGINEER
Westland Helicopters Limited, Yeovil,
Somerset, England. BA20 2YB

INTRODUCTION

The approach to loads monitoring is designed to fulfil two requirements. The first is the elimination, as far as possible of costly parts being thrown away because of overly conservative assumptions with respect to usage. The second is to assure the assumed safety standards by knowledge of the in-service aircraft utilisation.

Specific problems appropriate to the helicopter make this task more difficult, but nevertheless the fundamental steps have been taken to commence data acquisition.

A discussion of equipments with more limited applications are outlined.

LIST OF ABBREVIATIONS

ISA	International Standard Atmosphere
Hz	Hertz - Cycles/Second
S/N	Stress/Cycle
ASW	Anti-Submarine Warfare
VERTREP	Vertical Replenishment

THE ANALYSIS OF LOADS

(a) Problems with Helicopters

The helicopter dominant loads are created by the passage of each blade round the azimuth. The aerodynamic forcings cause higher frequency response that at once per rev, though rarely are these high frequency loads damaging in their own right.

Consequently the number of cycles relevant to a total analysis is very considerable, bearing in mind that a general main rotor frequency is about 3-6 Hz whilst tail rotors vary from 15-30 Hz.

Further, the high frequency signals can be important up to eight times the fundamental rotor frequencies requiring considerations of up to 50 Hz for main rotors and 240 Hz for tail rotors.

This being the case several corollaries follow:

(a) It is most practical to deal with peak to peak signals containing all the harmonics.

(b) Specific manoeuvres contain varying proportions of frequencies and are treated as a whole.

(c) Relatively small changes in aircraft attitude can affect the harmonics breakdown, which implies the measurement of a large number of manoeuvres.

As a consequence the procedure for establishing the loads for conditions are empirical, and have several pessimisms in the calculation methods discussed below.

(b) Standard Procedures

Generally the policy is to recognise a finite number of aircraft flight conditions. This number is of course naturally infinite, but is usally compressed down to 200-300 separable identifiable types. In previous early helicopters the number of conditions were as low as 30. In the obligatory development flight period these manoeuvres are flown with the objective of acquiring a sufficient number of each, to gain information on mean values and dispersion. At this point industry practice varies from applying a fixed factor for variability (as in WHL) or attempting to apply simple statistics. The flight condition is defined as the departure and return to steady state, say level cruise, and includes the transient effects of entry and recovery.

The variables involved not only include flight condition in terms of aircraft state, but also include altitude, temperature, centre of gravity, wind direction, gust

severity, external loads, weapon fits and the engagement of automatic flight control assistance.

This approach is emminently practical but is necessarily limited to the environment of the manufacturer which in our case is broadly at ISA conditions.

The affects of changing density with temperature and altitude are pronounced and therefore a more complete approach would be to take the aircraft to very hot and very cold environments, in the latter to include the effects of icing. The problems with moving the development exercise is that modern measuring systems have become highly computerised, using aircraft telemetry links in preference to on-board recording. This requires a very comprehensive receiving station and the capacity to record many hours of real time data on a very large number of data channels.

In these circumstances supporting an aircraft away from factory facilities becomes extremely expensive.

Nevertheless, trials have been conducted in Denmark, Norway and Spain, to pursue loads information, but reverting to on board data recorders, and a combination of manual and automatic analysis.

If we consider a typical spectrum we have as follows:

INBOARD DOGBONE PLAIN SECTION - LOADS AND DAMAGE RATES

10,050 LB, 5.6" FORWARD 0/2,000 FT

FLIGHT CONDITION	MEAN FLAP LOAD (LB IN)	MAXIMUM VIBRATORY FLAP LOAD (LB IN)	MEAN LOAD (LB IN)	MAXIMUM VIBRATORY LAG LOAD (LB IN)	DAMAGE/HR x 10
Take-Off	-11971	19830	856	6426	-
Hover	- 5545	18786	4196	7854	-
Rotor Engagement	-19741	18786	632	12423	2.212
Transition to Hover			-7024	13498	-
Spot Turn to Port	- 6778	16699	3533	7996	-
Spot Turn to Starboard	- 5457	18004	4047	9854	-
Transition from Hover	6188	15655	4842	7426	-
Sideways Flight	-13371	24005	4222	13423	39.514
Rearward Flight	-10195	22178	3670	13281	2.376
Climb	8476	10521	9165	8239	-
Partial Torque	5412	10656	-4841	7529	-
Autorotation Entry			16150	11815	-
Autorotation Recovery			5471	17578	-
Autorotation Steady	725	8903	-7196	6251	-
30 Kts Level Flight	5151	6879	1769	8097	-
Level Flight 0.4 VNE	3698	12716	-1945	5965	-
0.6 VNE	5241	10193	875	6965	-
0.7 VNE	5889	9762	2194	7705	-
0.8 VNE	6453	9886	3451	8604	-
0.9 VNE	7935	10564	4646	9664	-
1.0 VNE	7334	11796	5780	10884	-
30° Bank Turn 0.4 VNE	4000	17507	-1000	5949	-
0.6 VNE	6000	13776	0	6965	-
0.8 VNE	7000	13202	2000	8561	-
0.9 VNE	7000	14350	3000	9577	-
Once Per Flight					6.359
			TOTAL		50.461

If moreover we consider a typical strain gauge and instrumentation data gathering fit, and the spectrum applies to each component, some idea of the magnitude of the task is gained.

Westland 30 Aircraft Strain Gauge Standard

Main Rotor Hub Window and Counterbore

		Mean and Vibratory
Main Rotor Hub "Window" Gauge	F	M + V
Main Rotor Hub "Window" Gauge	G	M + V
Main Rotor Hub "Window" Gauge	H	M + V
Main Rotor Hub "Window" Gauge	I	M + V

		Mean and Vibratory
Main Rotor Hub "Window" Gauge	J	M + V
Main Rotor Hub "Counterbore" Gauge	U (UPPER)	M + V
Main Rotor Hub "Counterbore" Gauge	M (MID)	M + V
Main Rotor Hub "Counterbore" Gauge	L (LOWER)	M + V

Main Rotor

Hub Torque	(M000T)	M + V
Hub Bending Moment	(M000M) x 2	M + V
Flap BM at	(M032F) x 4	M + V
Flap BM at	(M068F)	M + V
Lag BM at	(M068L)	M + V
Flap BM at	(M142F)	M + V
Lag BM at	(M142L)	M + V
Flap BM at	(M196F)	M + V
Lag BM at	(M196L)	M + V
Flap BM at	(M310F)	M + V
Lag BM at	(M310L)	M + V
Flap BM at	(M480F)	M + V
Lag BM at	(M480L)	M + V
Flap BM at	(M640F)	M + V
Lag BM at	(M640L)	M + V
Torsion at	(M700T)	M + V
Flap BM at	(M810F)	M + V
Lag BM at	(M810L)	M + V
Lag Damper Load	(M Lag Damper)	V
Spider Arm Bending	(M SPAB)	M + V
Fore/Aft Cyclic Jack Load	(MFCC)	M + V
Lateral Cyclic Jack Load	(MLCC)	M + V
Collective Link Load	(MCLL)	M + V

Tail Rotor

Tail Transmission Torque	(T.DS.Tq)	M + V
Spider Arm Bending	(T.SP.AB)	M + V
Pitch Change Lever Load	(T.PCL)	M + V
Gearbox Shaft Bending	(T.GBS.B)	M + V
Flap BM at 20.8% Radius	(T.208.F)	M + V
Lag BM at 20.8% Radius	(T.208.L)	M + V
Flap BM at 33.1% Radius	(T.331.F)	M + V
Lag BM at 33.1% Radius	(T.331.L)	M + V
Flap BM at 50% Radius	(T.500.F)	M + V
Lag BM at 50% Radius	(T.500.L)	M + V
Flap BM at 50% Radius	(T.600.F)	M + V
Lag BM at 60% Radius	(T.600.L)	M + V
Flap BM at 70% Radius	(T.700.F)	M + V
Lag BM at 70% Radius	(T.700.L)	M + V
Torsion at 45% Radius	(T.450.T)	M + V
Torsion at 55% Radius	(T.550.T)	M + V
Torsion at 65% Radius	(T.650.T)	M + V

Airframe Transducers for G-BKKI

		Position No
1. Co-Pilot's Feet	- vertical	1
2. Pilot's Feet	- vertical	2
3. Co-Pilot's Seat Aft Outboard Attachment	- vertical	5
4. Pilot's Seat Aft Outboard Attachment	- vertical	6
5. Rear Cockpit Bulkhead	- lateral	7
6. Rear Cockpit Bulkhead	- fore and aft	8
7. Cabin Floor Port Side Approx St'n 1710F	- vertical	9
8. Cabin Floor Starboard Side Approx St'n 1710F	- vertical	10
9. Cabin Floor Port Side Approx St'n 450F	- vertical	11
10. Cabin Floor Starboard Side Approx St'n 450F	- vertical	12
11. Cabin Floor Port Side Approx St'n 1210A	- vertical	15
12. Cabin Floor Starboard Side Approx St'n 1210A	- vertical	17
13. Intermediate Tail Rotor Gearbox	- lateral	
14. Intermediate Tail Rotor Gearbox	- vertical	
15. Intermediate Tail Rotor Gearbox	- fore and aft	
16. Tail Rotor Gearbox	- lateral	

Position No

17. Tail Rotor Gearbox - vertical
18. Tail Rotor Gearbox - fore and aft

Engine Transducers for G-BKKI

1. Starboard Engine Power Turbine - lateral
2. Starboard Engine Power Turbine. - vertical

Additionally shown in the spectrum are the relative damage numbers in terms of Miner's Law from each of the conditions described for the specific component. This analysis shows how the design is generally driven up in strength until only a small proportion of high amplitude, short duration conditions cause fatigue damage. This can be readily seen since only a few conditions shown are leading to a calculated safe life of $\sim$5000 hours. As a result the amplitude AND persistance of each damaging condition are of great significance. (See Figure 1).

Furthermore for the helicopter, the cycle of running up the rotor, taking-off flying and landing, itself causes fatigue damage, and therefore the frequency of this characteristic in its own right is critical.

The reason for this characteristics in rotor systems is that in order to produce a practical design, joints are incorporated in the blade/rotor interface. The centrifugal load on these joints at nominal rotor speeds generates stresses, due to stress concentrations, greater than the endurance limit of the system. This in turn means that the component inevitably has a limited fatigue life related direclty to the number of rotor starts. Usually life factors and empirical tests are used to define safety margins in these components, but in principle the operational data requirement is the same.

Because Miner's Law cumulative damage rule contains only linear functions of applied cycles the implications is that if we assume only one condition is damaging and leads to the promulgation of a safe life, doubling the occurrence will halve the life, though in real terms it will mean that the probability of failure will rise above the accepted criteria after half the promulgated life has been consumed and only <u>may</u> lea to failure within the full life.

If the assessment of amplitude is incorrect then worse than linear effects can occur because of the shapes of the S/N curves and this emphasises the need fon comprehensive flying, or in-service loads measurement. (The S/N curve shapes are usually based on relevant coupon type data. The component mean strength is based on fatigue test results and on the production component).

Unlike the fixed wing conditions of flight, the blade of a helicopter is continually in a very disturbed airflow, and the classical manoeuvres of say pull-ups and turns are not as significant. Indeed, a non-agile civil helicopter safe life may be totally controlled by the transient entry and recovery to low speed sideways or rearwards flight.

(c) Types of Usage

The helicopter derives its usefulness from its flexibility; spectra are generally derived from either:

(a) Theoretical flight profiles.

(b) In-service user surveys, of various types.

(c) A combination of both.

The former are necessarily based on previous experience and can be erroneous for a new operator. The latter are usually restricted to periodic pilot questionaires which can fairly inaccurate. A Typical questionaire is shown in Table 1.

The usage may be detailed in a variety of roles typical of which are; ASW, SAR, VERTREP, CAPAD, Shorthaul Commuter, Long Haul, Ferrying etc.

Each of these are then broken down into a range of flight conditions (Table 2) based on heavy handed estimates of duration of flight.

The difficulty with this approach is that it does not interface with the fatigue analysis of the components and indeed for a new aircraft cannot be, and is consequently only a guide. Nevertheless, the measured aircraft loads from development flying are applied to these guides to formulate the safe life.

When operational patterns have been established attempts by use of questionaires may be made to refine the more tentative areas. Aircrew generally dislike 'excess' paperwork and it follows that useful answers will be in inverse proportion to the number of questions asked. The approach is further hampered by the very limited instrumentation of the standard in-service aircraft, and the pilot's notoriously inaccurate judgement of time. This inaccuracy stems from the length of time spent in high workload environments, quite understandably, and time seems very long when negotiating difficult manoeuvres such as

landing back on board a heaving ship's deck, or oil rig platform in gusty conditions. This tends to lead to overly pessimistic estimate of time in condition.

Regrettably it is precisely these areas which tend to be critical.

Examples of more detailed searching for knowledge of time spent in critical condition areas are shown in Table 3 for low speed SAR training.

On receipt of such information the designer is inclined to despair, and responds by either generating a brand new spectrum on the basis of the data, and then re-analysing the entire aircraft for safe life, or creates compositions of existing spectra adjudged to give the appropriate operational severity. The latter is more popular from the customer's viewpoint as it represents a much cheaper alternative.

All the foregoing emphasises the need for accurate data, on which to base judgement and assess effects.

(d) Operational Effects

Rotor Systems

For main rotor systems, all up weight, centre of gravity position, incremental 'g' force, and control inputs in terms of amplitude and rate all affect the rotor loads generated. As a result, the acquired data to make sense of the inputs needs to be fairly comprehensive.

Broadly, neither cg or all up weight have a pronounced effect on tail rotor components. They will normally suffer most from low speed manoeuvres, habitually prone to considerable scatter. Amplitude and rate of yaw input are the most critical parameters.

Transmission

For transmission systems, depending on type, broadly only torque demand causes problems, but where in aircraft like Lynx, the gearcase is used as the rotor mounting on the fuselage, the inputs for rotor systems come into play.

Structure

The structure of a helicopter is the area most like fixed wing, and broadly can be treated in a similar manner; however structural resonances are more common and can require investigation in their own right.

Undercarriage

The helicopter undercarriage is very different from the fixed wing type in that the whole design aim is increased energy absorption, it being considered that the vast percentage of helicopter landings will be vertical.

This exacerbates the problem of aircraft bounce in the run-on landing situation, but in either case the information controlling the life is obviously only the landing. The variables are aircraft attitude and rate of descent.

HODR - Helicopter Operational Data Recording

Clearly it would be desirable to instrument, comprehensively, each aircraft in the fleet and record all the data so as to individually life, safely, each aircraft; obviously at the present stage this is not possible.

As will be discussed later however, the start of this process is being attempted, but for simplicity we can break down the lifing task into three phases:

(a) Find out the time spent in manoeuvres.

(b) Find out the loads.

(c) Calculate the life for each critical section.

Ideally (a) and (b) should be done on the same aircraft but HODR attempts to evaluate (a), with a sample of the fleet and a home based aircraft to identify (b).

At present WHL have been involved (with RAE Farnborough) in the instrumentation and reduction of data from two in-service aircraft. A Mk 2 Sea King involved in pilot training, and a Mk 4 Sea King are currently flying with 'parameter' instrumentation.

The objective of this 'parameter' instrumentation fit is to be able to interpret the aircraft behaviour and allocate a duration to the recognised manoeuvre. With this accomplished for all the recognised manoeuvres a spectrum can be generated, and for unrecognisable situations the home based aircraft flown to attribute loads to this previously unanticipated area.

This 'parameter' standard includes the following:

Typical Sea King Parameter List (HODR)

Aircraft Motion Parameters

Normal g - main rotor centre line port side of cabin
Normal g - main rotor centre line starboard side of cabin
Normal g - main rotor centre line aft cabin
Lateral g - main rotor centre line aft cabin
Longitudinal g - main rotor centre line
Aircraft pitch attitude
Aircraft roll attitude
Aircraft heading ('yaw attitude')

Aircraft Flight Control Parameters

Collective Control Position) Future installations will also
Cyclic Control Lateral Position) include control servo positions
Cyclic Control Fore/Aft Position)
Rudder Pedal Position
Engine Torque No. 1
Engine Torque No. 2
Tail Rotor Torque
Main Rotor Speed
Indicated Airspeed
Pressure Altitude
Doppler Velocities

Strain Gauges

Tail Pitch Control Load)
Airframe Strain Gauges (2 off)) Installed on latest aircraft
Tail Pylon Hinge Strain Gauges)

Status Signals

Aircraft on Ground Event (Landing Gear Switch)
AFCS Engaged Event
HF Radio Transmit Event
Elapsed Time

The recording system starts with the release of the rotor brake and stops twenty seconds after application of the brake.

Further work is programmed to extend this information to five further Sea King and six Lynx.

Commonality of instrumentation between Sea King and Lynx has been achieved. A small cassette recorder is used to store multiplex data from the above aircraft systems signals and specially installed transducers via a programmable data logging system.

These signals are in the main low pass filtered at about 1.5 Hz. In addition some strain gauges are installed to measure strains directly in certain habitually critical areas.

The sample rate of these parameters is generally about 4-8 times per second, which is sufficient to describe control inputs and motion response of the aircraft. Strain gauges have to be sampled at much higher rates depending on the frequency of loading, as described in the introduction. The tape capacity gives a duration of about two hours and the equipment is described in Reference 1.

In order to reduce the volume of the strain gauge type of data an alternative recording method is currently under investigation. This process involves both low and high pass filtering of the signal and recording on separate channels the low pass filtered signal which is equivalent to the steady part of the signal, and the high pass portion, equivalent to the peak amplitude. Their time history and association is maintained such that at a later stage the stress analysis can be carried out, by replaying essentially tabulated data and calculating fatigue damage on this basis.

HELICOPTER OPERATIONAL DATA

The parametric data is processed by two basic methods:

(a) 'Automatic'

Computer reduction of the data is carried out to produce tables of specific information such as number of rotor starts, take-offs and duration of each sortie and counts of the number of autorotations. Other information is also gained such as engine torque spectra, airspeed versus altitude and airspeed versus angle of bank spectra. Aircraft lateral and normal accelerations are found by combining the accelerometer signals.

This type of processing of the parametric data will, it is hoped, provide usage spectra representative of the various aircraft roles for comparison with the assumed spectra and such old standards as CAM6. Typical Sea King spectra are included in Reference 2. Cycle counts and damage analysis will be made on strain gauge data.

(b) 'Manual'

Time histories of the parameters are displayed for visual examination to discover any unexpected manoeuvring of the aircraft which might have a potentially damaging effect. This process is at the moment of necessity visual (using engineers with flight experience) because at the present time the computing power and software requirement for automatic manoeuvre recognition are prohibitive.

The technique employed is to look for occurrences of high 'g' loading and/or rapid change of aircraft attitude which imply high structural or dynamic component loads or high rotor moment. The manoeuvres are then described as precisely as possible for the inclusion in the flight programme of a load survey helicopter or for insertion in the assumed flight spectra for the mark of helicopter. On the one aircraft on which this work has been carried out it has been found difficult to identify low speed manoeuvres but future aircraft data acquisition systems will include doppler velocities and tail rotor torque information to assist in this area.

CONCLUSIONS

The approach centres on answering the many questions of what does an aircraft in the field actually do? The problem with this question is that it is never fully answered in that an individual machine may indeed be suffering the worst loads because of topographic problems, or lack of maintenance personnel, and happen not to be the one selected for analysis.

The state of the art with respect to usage is so coarse however that the seeding of service aircraft across the operators is likely to be invaluable in identifying the general trends in critical areas to put us in a much stronger position than today.

The benefits of such knowledge can be either the avoidance of catastrophic underdesign, or the recognition of over pessimism in the lifing policy. On the assumption that we will not have the former, the monetary benefits of the latters could be staggering.

The effect of longer safe life and the consequent saving of spares purchase and inventory could replay the cost of the loads measurement many times over, and it is hoped it will.

The programme for the Sea King flight on the data gathering aircraft is taking place at present. The data acquisition in-service is now starting to become available.

From the Mk 2 aircraft data from the first HODR programme, the basic results yield the following typical data:

(a) Bar charts of Indicated Airspeed and altitude in bands of speeds in excess of 40 kts. This converts into percentage of time at speed and altitude to compare with spectral assumptions.

(b) Counts of take-offs per hour.

(c) Occurrence of autorotation.

(d) Summations of twin engine powers and times.

(e) Rotor speed values and occurrence.

(f) Roll angle amplitude and occurrence leading to a first estimate of time in turns.

From early analysis of the above some early trends are discernable most of which support the view that the spectral assumptions are pessimistic and that fiscal savings may be made. Two of the most obvious effects which have already been noted, which could have adverse life effects are that:

(a) The amount of autorotation is higher than anticipated which increases the torque cycling of the gearbox.

(b) Flight durations are shorter than anticipated, which adversely affects components affected take-offs, and rotor starts.

Limited Aircraft Fit Devices

Whilst the HODR programme attempts to look at overall aircraft parameters, several attempts, of which two will now be highlighted, have been made to address areas which are more controllable, in that either instrumentation already exists than which loading information can be extracted, or the objective of measurement is strictly limited.

Health and Usage Monitor - Westland 30

Torque Exceedance

The loads in the main transmission system of the Westland 30 are dominated by the utilisation of torque in the single and twin engine mode. If knowledge of the system torques could be acquired it would be

possible to calculate the effect on the individual gears. The bearings are controlled by a health monitoring package of oil analysis.

As part of a similar aim on the engine an automatic processor with enhanced software capacity has been procured and is currently under test.

The device fitted to the aircraft is shown in Figure 2 and performs the following functions:

(a) Engine Power Monitoring.

(b) Engine Usage Monitoring.

(c) Transmission Torque Monitoring.

(d) Transmission Usage Monitoring.

(e) Rotor Torque Usage Monitoring.

To perform these calculations for the gearbox the unit uses the torque for each engine.

The total range of torque between zero, and twice maximum continuous is banded, into nine bands. The system recognises the band in which the torque lies, and sums the time spent in each band. It conducts the same calculations for Engine No. 1, Engine No. 2 and the total torque from Nos. 1 and 2 including the effect of torque split. (See Figures 3 and 4).

The data is retained in updated stores of torque and time. For rotating gears it is not generally necessary to know the sequence effect of torques because each gear naturally has a large number of teeth. The aircraft torque is relatively slow moving function compared to the tooth passage frequency. Consequently, each tooth on a gear sees a relatively long duration of constant root bending cycles in each band, corresponding to a miniature constant load amplitude application.

Therefore as long as the sequence effects are included on the transmission fatigue test, and WHL commonly use a programme load approach on new gearboxes, it is not necessary to know the applied sequence of stresses.

The fatigue S/N curve for each gear in the gearbox is included in the system memory on the look-up table, and at the end of a flight the stored data on torque and time is applied to the fatigue damage algorithm to calculate a 'damage index', for each component. When the 'damage index' equals unity for any component, the affected item is life expired. The unit is interrogated at a pre-determined frequency, but has been designed to have sufficient storage for 1000 hours.

This system effectively puts the gears on-condition with respect to life, and as long as the record of life used is kept with the gear, this condition can be maintained until life expiry on the most critical unit.

With service experience, it is expected that the aircraft operator should be able to look back on the growth of damage index with flight hours for his operation(s), and with system familiarity be able to predict a scheduled removal from the aircraft.

Since the other components within the transmission are monitored by oil condition one way or another, as previously mentioned, the entire main transmission system could become on-condition.

The device is currently undergoing system proving trials at WHL, but should shortly be fitted to a development aircraft for flight trial purposes.

There is no reason why the tail transmission system could not be similarly treated, and the later variants of the Westland 30, are fitted with a solid state tail torquemeter precisely for this purpose. This data is added to an enhanced, but similar computer package, with this aim in mind.

Rotor Head Monitor

Both Lynx and Westland 30 are fitted with a common semi-rigid rotor head. The high control power available implies that small control movements give high rates of response, and correspondingly high loads. One of the areas in which this is a potential problem is is in the area of aircraft taxying.

At the present time detailed instructions are written into the pilot's manual describing the procedure for control inputs in the taxying mode. The particular problem areas are in operation from poorly prepared sites and grass landing strips.

The problems comes about when trying to start off the aircraft in soft conditions. The pilot wishes to create a forward motive force. The method of doing this is to tilt the thrust vector forward by applying cyclic stick movements. As a result he also applies main rotor head moment. Depending on the softness of the ground, these loads can be quite large.

Large airports, or controllers of airspace, can ban hover taxying, demanding that the helicopter behave as if it were a fixed wing aircraft. This imposes an unusual, and generally unknown constraint on an aircraft not designed for prolonged use in this way.

The procedure would be quite straightforward if a head moment gauge could be fitted for the pilot, such that he could curtail operations within the acceptable stress envelope. This requires rotational information from strain gauges on the main rotor head to be brought down to the non-rotating co-ordinate system of the cockpit.

In development aircraft this is done by a mechanical highly complex slip-ring whose reliability is not good. This problem of serviceabilty becomes particularly acute in the long term, and is one of the major difficulties of in-service data acquisition. However, a radio slip-ring using multiplexed signals transmitted over a short distance using RF signals is currently under development by WHL. The amount of information which can be transmitted is limited but it is nevertheless ideal for this application.

The strain gauge signals for two adjacent aims of the four bladed design are added to give spacial vector of moment by suitable filtering. The steady component of lift is also extracted to allow for the mean stress on the component, and this data is supplied to the pilot on a special cockpit gauge.

By keeping within a set of pre-determined boundaries, the generation of inadvertent loads causing damage is precluded. In this way the promulgated life is assured and possible spectral exceedances curtailed.

These pursuits are extended to a whole health monitoring package (Reference 5) on Westland 30 Series 300. Figure 5 shows a completely interfaced package covering a great deal more of the system than those related only to fatigue sensitive loads directly.

Main and tail torques are recorded in an advanced package similar to Westland 30 Series 200. Main rotor head usage is extended into areas other than but including the taxi mode.

Engine power checking and lifing, oil condition, and selective vibration analysis add together to provide as large an on-condition assessment as possible all of which is designed to eliminate the unknowns in user operation and spectrum.

CONCLUDING REMARKS

The HODR task in collaboration with MOD is the most complete approach possible at this time, to determine the utilisation, and acquire loading data. The nature of the task requires considerable aircraft management to ensure that as many role configurations are covered as possible. Since the data is acquired in real time, the acquisition of a statistically significant amount will be spread over the next few years. Nevertheless, early signs indicate that a considerable number of improvements will be accrued from even early data received.

This approach also provides the ability to encompass new roles for an in-service aircraft type, in a much more rigorous way than current estimating methods.

The methodology should also provide an information database against which to test loading theories (such as random load) and extend statistical scatter knowledge on times.

REFERENCES

Reference 1 - 612/SA/36580/001/Issue 2
HODRS System - Plessey plc

Reference 2 - Stress Note SN 227 (Flight Condition Spectra for Fatigue Substantiation)

Reference 3 - WK/EA/034 - Specification for Health and Usage Monitoring Equipment

Reference 4 - Health and Usage Monitoring in Helicopters
D G Astridge and J D Roe (Authors)
Stressa, Italy

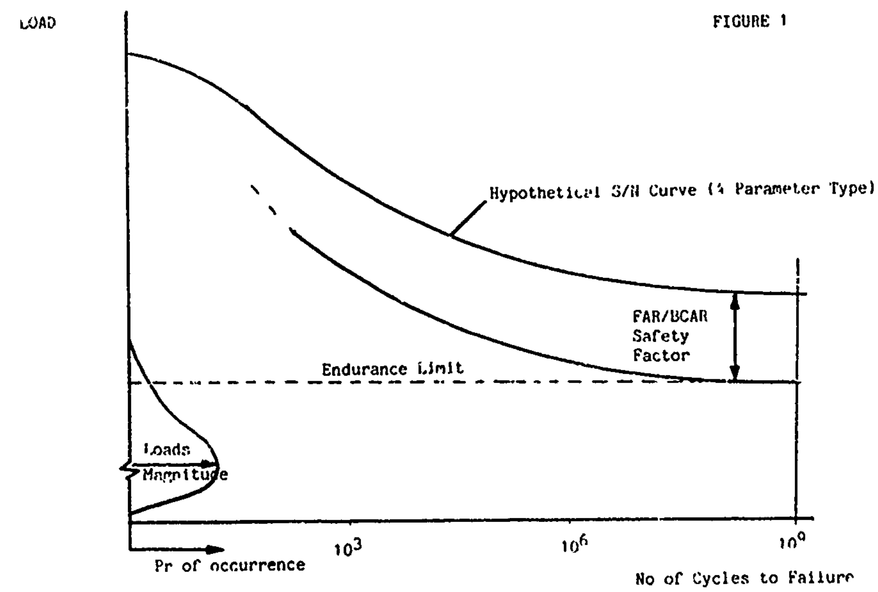
Reference 5 - WER 142-30-00348

Reference 6 - Health Monitoring of Helicopter Gearboxes
D G Astridge
Aix En Provence

Note all the references and internal documents are available from the author at Westland Helicopters Limited, Yeovil, Somerset, England.

LIST OF FIGURES AND TABLES

FIGURE 1


LOAD
Hypothetical S/N Curve (4 Parameter Type)
FAR/BCAR
Safety
Factor
Endurance Limit
Loads
Magnitude
Pr of occurrence
10^3
10^6
10^9
No of Cycles to Failure

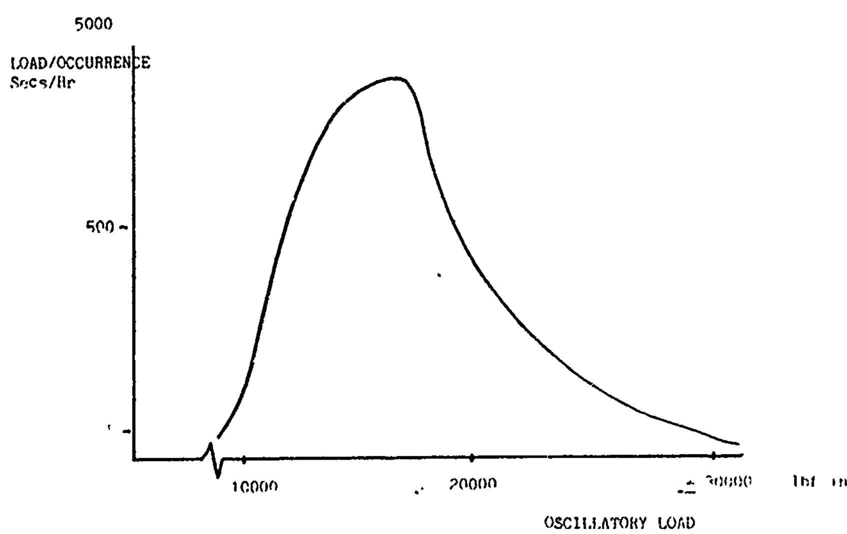

5000
LOAD/OCCURRENCE
Secs/Hr
500
10000
20000
± 30000
lbf in
OSCILLATORY LOAD

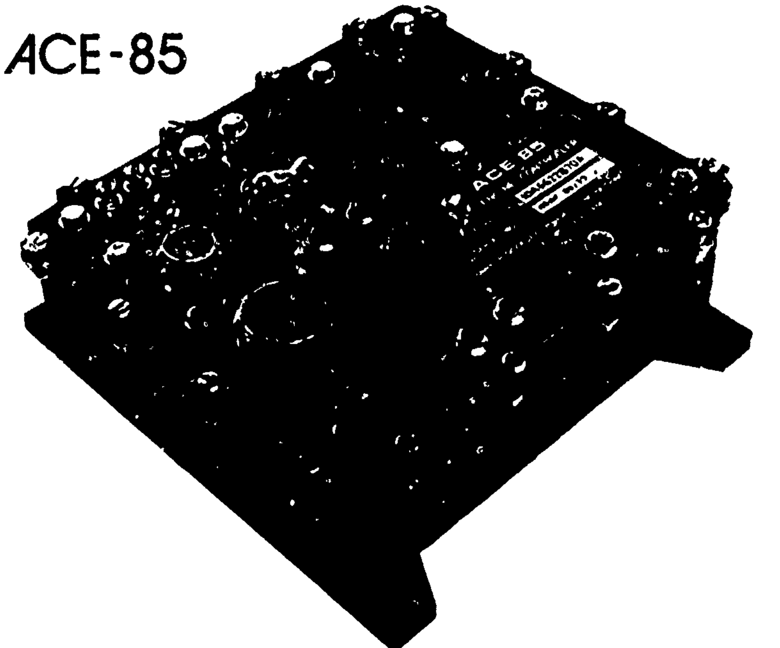
FIGURE 2

hsde

DIGICON®

MICROPROCESSOR CONTROL SYSTEM

ACE-85



- High integrity/self-checking software
- In-built non volatile memory
- Serial I/O for diagnostic and on-line data exchange
- 1 amp outputs for actuator and relay drives
- Low power consumption and heat dissipation
- Analogue back-up overspeed/over-temperature control
- In-built LP or HP pressure transducer
- Growth capability on spare PCB within unit

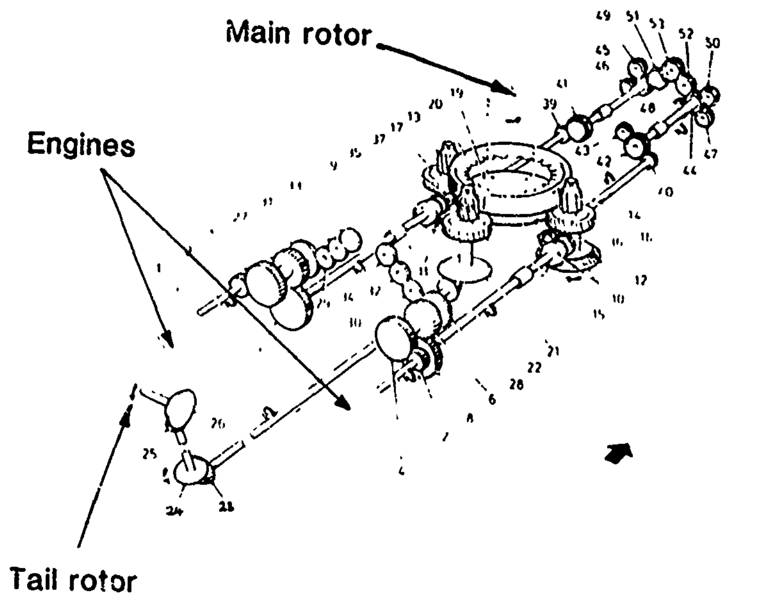
The ACE-85 range of full-authority general-purpose digital engine controllers provide the user with a powerful computer based on the INTEL 8085 microprocessor and the twenty years of control experience by HSDE.

The high performance design contained in the rugged environmentally protected unit is capable of meeting the most stringent requirements that today's industrial engines and aircraft gas turbines demand.

The high reliability and integrity is achieved by using well established quality components along with specially developed self-checking software. Modular construction using multi-layer p.c.b. techniques and film wire inter connections contained within an aluminium alloy cast housing contribute to the integrity, maintainability and overall low cost of the unit.

FIGURE 3

LYNX/WESTLAND 30 TRANSMISSION SYSTEM



GEAR	TITLE	GEAR	TITLE
1	Input Pinion	28	Layshaft Fan Drive Gear
2	Input Pinion	29	Fan Drive Idler Gear
3	Layshaft Wheel	30	Fan Drive Idler Gear
4	Layshaft Wheel	31	Fan Drive Idler Gear
5	Layshaft Pinion	32	Fan Drive Idler Gear
6	Layshaft Pinion	33	Fan Drive Gear
7	Output Gear	34	Fan Drive Gear
8	Output Gear	35	Accessory Driver, Aft
9	Bevel Driver M G B Input	36	Accessory Driver, Aft
10	Bevel Driver, M G B Input	37	Accessory Driven, Aft
11	Bevel Driven, M G B Input	38	Accessory Driven, Aft
12	Bevel Driven, M G B Input	39	Accessory Driver, Fwd
13	Conformal Pinion	40	Accessory Driver, Fwd
14	Conformal Pinion	41	Accessory Driven, Fwd
15	Conformal Pinion	42	Accessory Driven, Fwd
16	Transfer Spur - Driver	43	Tacho, Spur
17	Transfer Spur - Driven	44	Freewheel HSG, Spur
18	Transfer Spur - Driven	45	Freewheel HSG Spur
19	Transfer Spur Idler	46	Hydraulic Pump Spur
20	Conformal Wheel	47	Hydraulic Pump Spur
21	Bevel Driven T T O	48	Oil Pump Spur
22	Bevel Driven T T O	49	Generator Spur
23	Inter G B Output	50	Generator Spur
24	Inter G B Input	51	Idler Spur
25	Tail G B Output	52	Idler Spur
26	Tail G B Output	53	Idler Spur
27	Layshaft Fan Drive Gear		

FIGURE 4

WESTLAND 30 TORQUE MONITORING SYSTEM

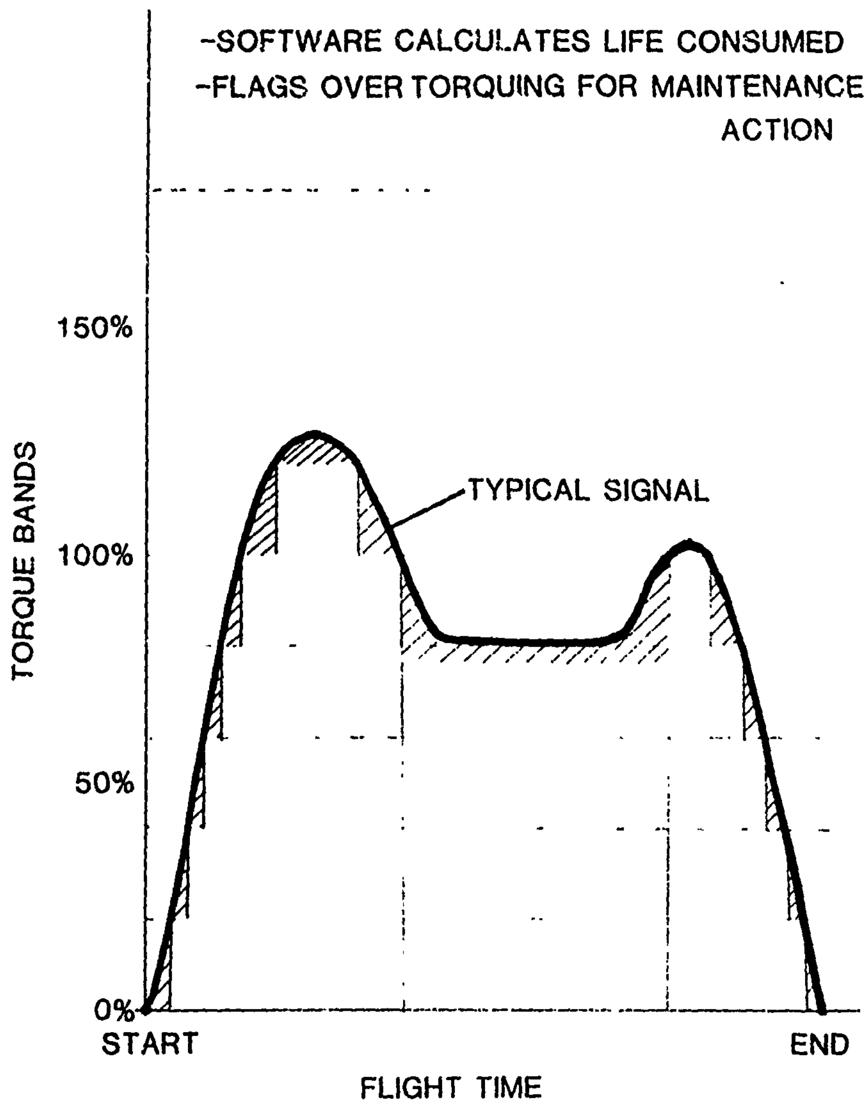


table 1

MOD FORM 725 (HELICOPTER) (PPOV-1983)

TAIL NO	BASIC AC WT	SHEET NO	AIRCRAFT TYPE

AIRFRAME HRS	MONTH HRS	LANDINGS/MTH		CGI	OSF	Yell	Red	GUST	OSF		AUW	Total	Ext Load	Int Load	Aft CG	> 100kt
		ROTOR S/S/MTH			O				O		A					
					S				S		B					
					F				F		C					

CAPTAIN	DATE(DAY:MTH)		NO OF [illegible]DINGS		CGI	YELLOW	RED	GUSTS	START AUW	
	SORTIE DURATION		ROTOR START/STOPS						END AUW	

LOAD UPTAKES	1	2	3	4	5	6	7	8	BALANCE OF SORTIE (UNLOADED)	
INTERNAL LOAD										
EXTERNAL LOAD										
AUW DURING CARRIAGE										
DURATION OF CARRIAGE	:	:	:	:	:	:	:	:	MEAN AUW	
AFT CG (✓ OR X)									DURATION	
TIME AT > 100 KT	:	:	:	:	:	:	:	:	AFT CG	
									TIME > 100	

table 2

PROFORMA

SEA KING MK 2/5

All times in Minutes

		1	2	3	4	5 (Mk 5 only)	6	7
a	Sortie Type	TRAINING GENERAL	TRAINING ASW	TRAINING IF	ASW	ASW 2	SAR	VERTREP
b	Sortie Duration hrs	1.00	2.00	1.00	4.00	4.00	3.00	2.00
c	Max AUM lb	19500	19700	19500	20500	21000	20000	21000
d	No. Rotor Starts	1	1	1	1	1	1	1
e	No. Landings	20	1	1	1	1	1	5
f	No. Decels or Flares to Hover	15	20	2	18	5	4	30
g	Total Time in Hover and Low Speed Manoeuvres	20 min	50 min	05 min	1.00 hr	20 min	15 min	50 min
h	Time Hovering into Wind (± 20° on Nose)	15 min	40 min	3 min	45 min	15 min	12 min	40 min
j	Time in Sideways Flight to Stbd	2 min	3 min	0 min	3 min	1 min	1 min	2 min
k	Time in Sidways Flight to Port	2 min	3 min	0 min	3 min	1 min	1 min	2 min
l	Total Time in 'Transit' Flight	10 min	15 min	15 min	1.00 hr	1.10 hr	2.00 hr	5 min
m	Speed kts	90 kts	100 kts	90 kts	110 kts	110 kts	110 kts	60 kts
n	Nominal Altitude ft	1000	750	3500	1000	2000	1000	500
p	Time 'on Station'	50 min	1.45 hr	45 min	3.00 hr	2.50 hr	1.00 hr	1.55 hr
q	Average Speed kts	50 kts	60 kts	60 kts	60 kts	75 kts	60 kts	60 kts
r	Nominal Altitude ft	500	200	3500	200	2000	1000	200
	Remarks and Additional Details ie Number Sonar Dunks 'Sonobuoys' Wt of Underslung Load	q = Approx average between exercise speed and hover except column 7 which is the average speed for transit with an underslung load						

table 3

SEA KING MK 3

RECORD OF SIDEWAYS FLIGHT DURING DECK WINCHING EXERCISES

NOTE : It is very important that the speeds, times and weights you record below are those which apply ONLY to the pr[illegible]s during deck winching i[illegible] which you were in SIDEWAYS FLIGHT.

	TIME IN MINS SPEND IN SIDEWAYS FLT AT WEIGHTS		AVERAGE WIND SPEED	TOTAL TIME (MINS)/AV SPEED (KTS) FOR SIDEWAYS FLIGHT DURING:			
DATE	< 19,500 BUT > 20,500	< 20,500	KTS	STANDARD DECK	ODD DECK STBD	ODD DECK PORT	DOWNWARD ODD DECK
(i)	(ii)	(iii)	(iv)	(v)	(vi)	(vii)	(viii)
	Notes of explanation for WHL: Time at weights < 19,500 lbs is calculated by exception/			Aircraft moves in direction shown by diagram in particular column. = Wind Direction = Aircraft Track			

SESSION II - FATIGUE ASSESSMENT & COMPARATIVE RESULTS

SUMMARY RECORD

by
G R Graham
Department of National Defence
Ottawa, Ontario, Canada

The presenter of paper No 8 was asked if the seasonal variation in damage rates reported indicated that fin loading was primarily due to gusts; he agreed that this was the case. In response to a comment on the adequacy of the highest sampling rate used (28 samples/sec), the rate used was subject to the limitations of the data collection system available, and was therefore something of a compromise.

The discussion on the CF-5 vertical stabilizer programme covered the flight programme itself, the observed spectrum, and the relative sensitivity of strain gauges and the mechanical strain recorder (MSR). The test sorties were flown by a Canadian Forces test pilot who was familiar with squadron operations; he was instructed to exaggerate the use of rudder in the short-term trial in order to establish a conservative set of results. Comment was made that this instruction was omitted from the longer-term programme of flight trials and that the observed agreement between short and long-term trials was therefore a matter of chance.

It was observed that the fin load spectrum was not symmetrical. No special investigation had been mounted to establish the reasons for this phenomenon, but it was believed that turns in one direction were more frequent, for operational reasons.

The short term programme showed a greater sensitivity to small amplitude cycles, since the strain gauges fitted to the tail structure were more sensitive than the MSR response. No attempt was made to correct for this truncation effect. Short crack effects were assumed absent, and the Forman equation and Willenborg retardation model were used in analysis. In response to questions on the performance of the MSR, the author observed that the instrument performed quite well at the outset of the programme. However, as time went on the quality of the trace deteriorated, and difficulty was experienced in reading the tapes, together with appreciable processing delays and data loss.

Similar questions on experience with the MSR were directed to the author of Paper No 10. Here again, MSR results had not been good, certainly not as good as with European users. A US speaker commented that USAF is abandoning use of the MSR, because maintenance difficulties and the expected greater reliability of microprocessor-based systems.

A strain gauge-based system was not considered for the F-16, possibly because of the perceived limited reliability of strain gauges, but also because a relatively cheap system was required. Costs were kept down by recording parameters which were already measured by existing instrumentation. In subsequent discussion on the large number of parameters mentioned in various papers presented during the session, it appeared that almost all recorded parameters are used in some loads computation, although considerable scepticism was expressed in some quarters about the dependability and accuracy of such load predictions, taken across the complete flight envelope and for widely varying aircraft configurations.

Comment was made on the number of aircraft fin problems mentioned during the session, suggesting that there may be a deficiency either in design requirements or in quality of design for such structure. In reply, one speaker felt that the observed problems were more related to fatigue than to static strength. They appeared to stem either from inadequate load spectra or from changes in the operational environment for the aircraft in question.

The operational usage of new fighters is increasing the effects of unsteady aerodynamics. Questions were raised on what sampling rates were required to define adequately the load histories for major structural components. It was suggested that feasible rates were largely a function of the aircraft type being surveyed. Space is not a problem on large aircraft, and recording equipment with very high response rates can be installed. However, space limitations on small fighter aircraft significantly affect the capability of the equipment carried. Unfortunately, the requirements may be more pressing on the smaller, more agile aircraft.

The two helicopter papers showed a considerable commonality of view, reflecting the considerable difficulty in acquiring and correlating the large number of parameters needed, and the special problems of high vibration levels and transfer of data across rotating bearings. Comment was made that the US Army experience in South-East Asia indicated that helicopters are being subjected to loads outside the design envelope. The question was raised as to whether this situation was acceptable under war-like conditions, or should design requirements be re-evaluated?

In subsequent discussion it was agreed that pilots are bound to try and get the best operational performance out of their machines (whether rotary-wing or fixed-wing). Accepting that there will be a limited number of excursions beyond the nominal operational envelope, it appeared necessary to design with some form of durability allowance built into the structure, to allow for such usage. National practice varied somewhat here.

EVALUATION OF OPERATIONAL LOADS TO VERIFY STRUCTURAL DESIGN

by

H. Struck and H. Balke*
Load Criteria Section
Messerschmitt-Bölkow-Blohm, UTE243
Hünefeldstr. 1-5, D-2800 Bremen1

SUMMARY

A method of load evaluation derived from operational manoeuvres in addition to the design requirements applied will be presented. The method is based on the hypothesis that all manoeuvres trained and flown by the Air Force are standardizable. Some relevant parameters have been chosen that are suitable to describe the manoeuvre time history sufficiently with respect to load analysis.

For two fighter aircraft and a few manoeuvre types the standardization of manoeuvre parameters will be demonstrated. By means of standardized manoeuvres the correlated parameters necessary for deriving structural loads, including control surface deflections, are determinable.

Operational loads on main structural components have been evaluated by applying a manoeuvre model. In conclusion a comparison of extreme operational loads evaluated with the manoeuvre model and those determined by the design requirements (MIL-8861) will be given.

1. INTRODUCTION

Aircraft structures are designed in accordance with the relevant regulations and based on a philosophy defining the load level so as to cover all loads expected in service. No explicit mention is made of the correlation between design loads and loads in service. In practice, manoeuvres, especially combat manoeuvres, are flown in accordance with given, practiced rules that lead to a specified motion of the aircraft in space. This fact gives rise to the idea of analysing the manoeuvres and deriving loads from them.

In Germany, an evaluation of Combat-NATO-Manoeuvres is being made with the aim of deriving operational loads by analysing measured parameters in operational flights. These parameters include the time history of the aircraft response and the control deflections. The flights have been performed at the test centre of the German Air Force on two aircraft. The evaluation of the manoeuvres is sponsored by the Ministry of Defence and will be continued.

Within the scope of this evaluation, an attempt is made to find a way of load analysis from operational manoeuvres in addition to the applicable design specifications. The evaluation is based on the assumption that it should be possible to standardize the manoeuvres trained and flown by the Air Force.

This means in detail that it should be possible to find a standardized time history for each type of manoeuvre, which is independent of the extreme values of the relevant parameters. Based on this assumption, it was analysed how the evaluation of structural loads could be realized after previous standardization of manoeuvres.

2. STANDARDIZATION OF OPERATIONAL MANOEUVRES

The parameters of the aircraft motion should be chosen in a way that recording and evaluation cause minimum expense. This can be achieved by using parameters available from the gyro or other existing systems of the aircraft, for example the attitudes (Euler angles) and/or angular velocities (roll, pitch, and yaw rates). In this evaluation, 3 attitudes (angle of pitch, bank, heading) and the load factors (vertical and lateral) have been analysed with respect to the feasibility of deriving design loads in this way.

The question is, which are the response parameters to start with? Three possibilities were investigated:

Attitudes:		Angular rates:		Load parameters:	
angle of bank	ϕ	roll rate	p	load factor	n_z
angle of pitch	θ	pitch rate	q	load factor	n_y
angle of yaw	ψ	yaw rate	r	roll rate	p
				additional:	
				pitch rate	q
				yaw rate	r

*) under sponsorship of the German MOD
and Contract Nr. T/RF 43/.../A1413...C1451

Fig. 1 shows the flow diagram of the data analysis. With all three input sets it is possible to

- complete the aircraft response parameters
- derive control deflections
- determine structural loads

In this paper the load parameters as input have been applied.

Because several manoeuvres of the same type are different in amplitude of motion and in manoeuvre time, for a requisite comparison a two-dimensional normalization is necessary. In Fig. 2 the procedure of normalization is illustrated. The ordinate presents one of the parameters of motion ($y = \phi, n_z, p, \ldots$) for several manoeuvres of the same type ($Y_1, Y_2, \ldots$). These parameters are normalized by relating them to the maximum value which has occurred. That means the maximum value of each normalized parameter becomes $Y = Y_1(max) = Y_2(max) = 1.0$. The time is presented by the abscissa (t), whereby the executing manoeuvre time is marked by t_1 respectively t_2 for several manoeuvres. The normalization is accomplished in a way that

- firstly, the manoeuvre time is chosen as the value 1.0 ($t_1 = t_2 = T = 1.0$)
- secondly, the extreme values of the relevant parameters coincide at the same normalized time.

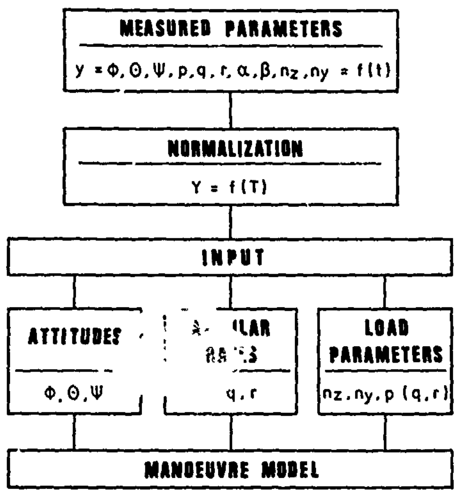


FIG.1 POSSIBILITIES OF ANALYSIS

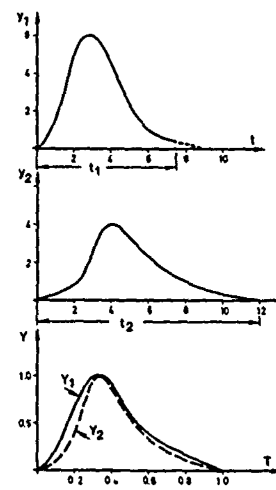


FIG.2 NORMALIZATION OF PARAMETERS

After normalization of the parameters measured the arithmetic mean values for all manoeuvres of the same type will be formed and corrected with respect to the steady state at the end of a manoeuvre. For demonstration of the normalized parameters and the formed mean values the results are plotted for the High-G-Turn manoeuvre in Fig. 3.1 - 3.8.

The mean values of all normalized parameters for all manoeuvres have been formed in combination with smoothing of the time history. For reasons of compatibility, the normalized data have to be tuned, that means the relation between Euler angles and angular rates is verified with the equations:

$$p = \dot{\Phi} - \dot{\Psi} \sin \Theta$$

$$q = \dot{\Theta} \cos \Phi + \dot{\Psi} \sin \Phi \cos \Theta$$

$$r = -\dot{\Theta} \sin \Phi + \dot{\Psi} \cos \Phi \cos \Theta$$

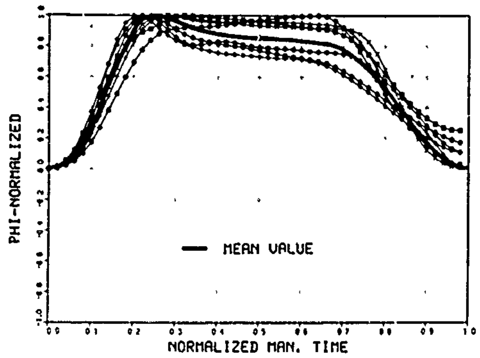


FIG.3.1 NORMALIZED PARAMETERS AND MEAN VALUES SEVERAL HIGH-G-TURN-MANOEUVRES

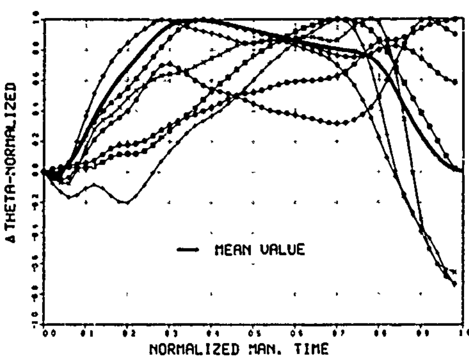


FIG.3.2 NORMALIZED PARAMETERS AND MEAN VALUES SEVERAL HIGH-G-TURN-MANOEUVRES

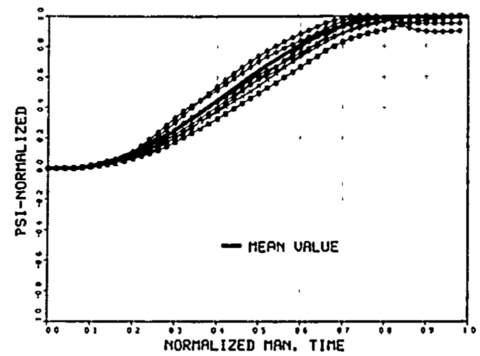


FIG.3.3 NORMALIZED PARAMETERS AND MEAN VALUES SEVERAL HIGH-G-TURN-MANOEUVRES

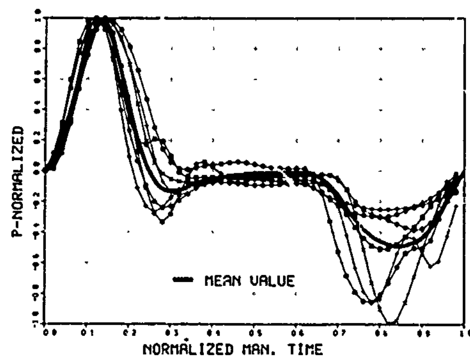


FIG.3.4 NORMALIZED PARAMETERS AND MEAN VALUES SEVERAL HIGH-G-TURN-MANOEUVRES

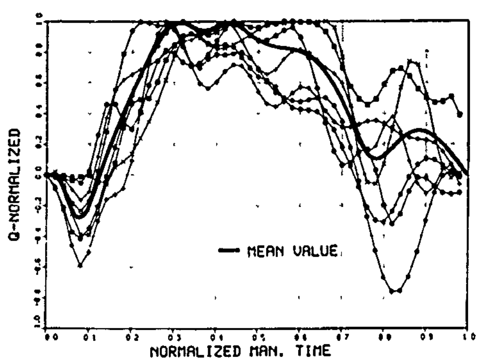


FIG.3.5 NORMALIZED PARAMETERS AND MEAN VALUES SEVERAL HIGH-G-TURN-MANOEUVRES

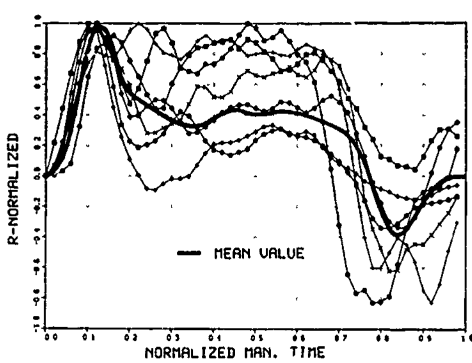


FIG.3.6 NORMALIZED PARAMETERS AND MEAN VALUES SEVERAL HIGH-G-TURN-MANOEUVRES

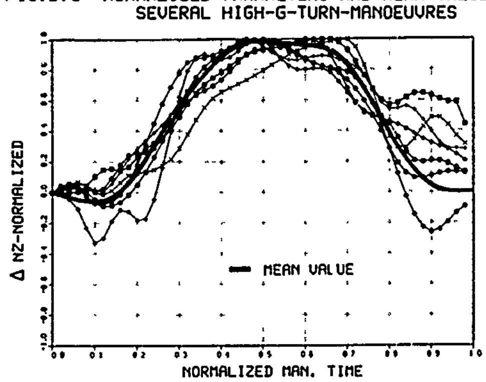


FIG.3.7 NORMALIZED PARAMETERS AND MEAN VALUES SEVERAL HIGH-G-TURN-MANOEUVRES

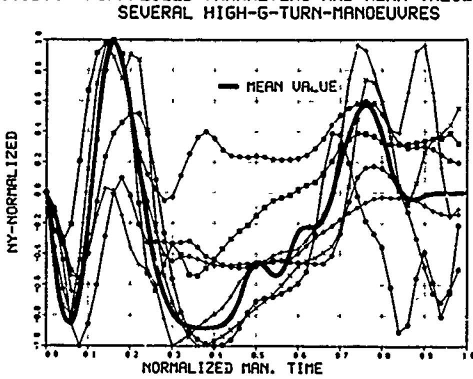


FIG.3.8 NORMALIZED PARAMETERS AND MEAN VALUES SEVERAL HIGH-G-TURN-MANOEUVRES

The result is the standardized manoeuvre containing the parameters:

- attitudes ϕ, θ, ψ
- angular rates p, q, r
- load parameters n_z, n_y

The standardization procedure is shown in Fig. 4. For two NATO-manoeuvres, the standardized parameters are presented: - High-G-Turn in Fig. 5.1 - 5.4
- High-G-Barrel-Roll o.T. in Fig. 6.1 - 6.4

This standardization procedure has been applied to a second type of aircraft for the same operational manoeuvres. For the High-G-Turn manoeuvre, the comparison of the main parameters has been plotted:

- symmetrical parameters n_z, q in Fig. 7.1
- roll parameters ϕ, p in Fig. 7.2
- yaw parameters ψ, r in Fig. 7.3

In general, good agreement for the relevant parameters could be found.

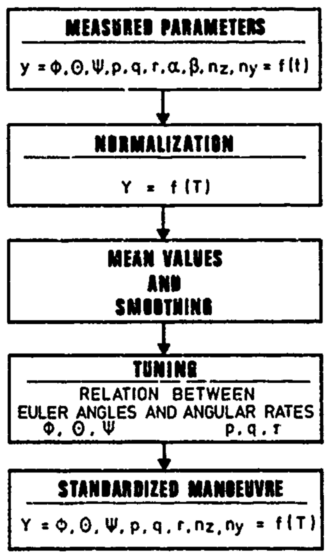


FIG.4 STANDARDIZATION OF OPERATIONAL MANOEUVRES

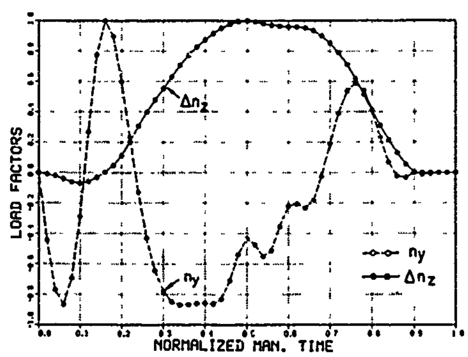


FIG.5.1 STANDARDIZED MANOEUVRE HIGH - G - TURN

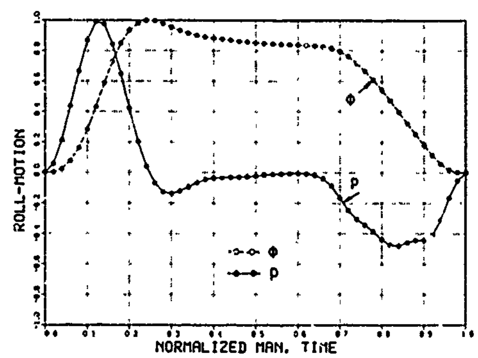


FIG.5.2 STANDARDIZED MANOEUVRE HIGH - G - TURN

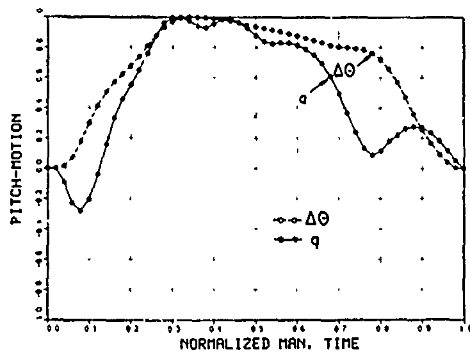


FIG.5.3 STANDARDIZED MANOEUVRE
HIGH - G - TURN

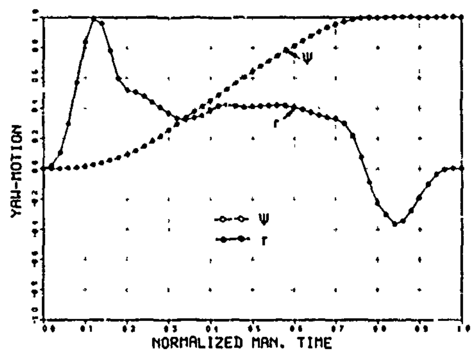


FIG.5.4 STANDARDIZED MANOEUVRE
HIGH - G - TURN

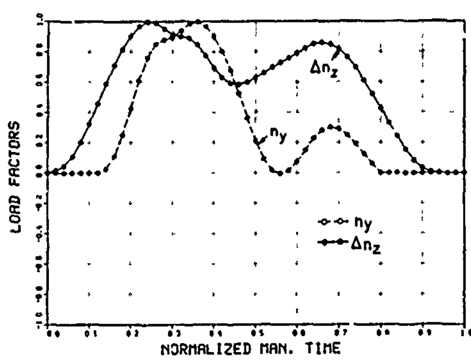


FIG.6.1 STANDARDIZED MANOEUVRE
HIGH - G - BARREL OT

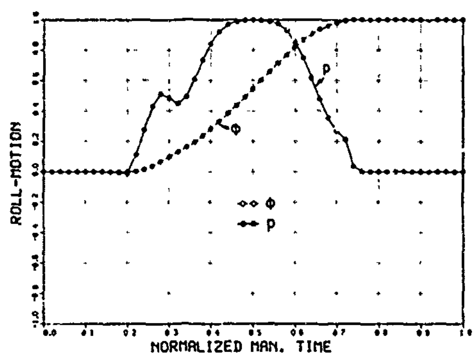


FIG.6.2 STANDARDIZED MANOEUVRE
HIGH - G - BARREL OT

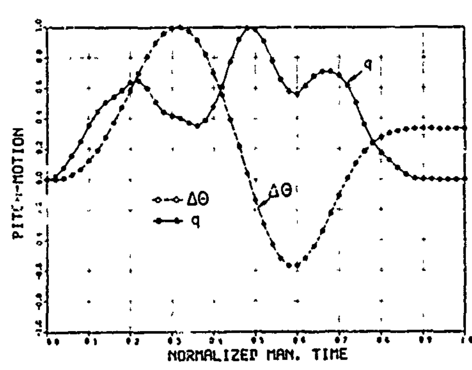


FIG.6.3 STANDARDIZED MANOEUVRE
HIGH - G - BARREL OT

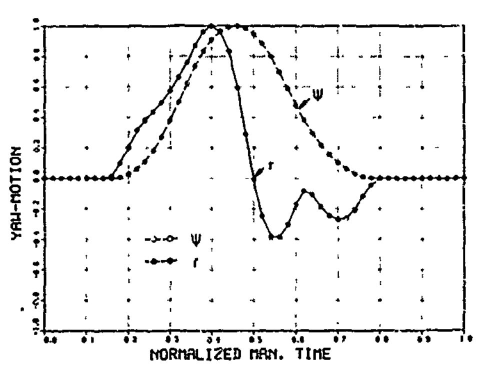


FIG.6.4 STANDARDIZED MANOEUVRE
HIGH - G - BARREL OT

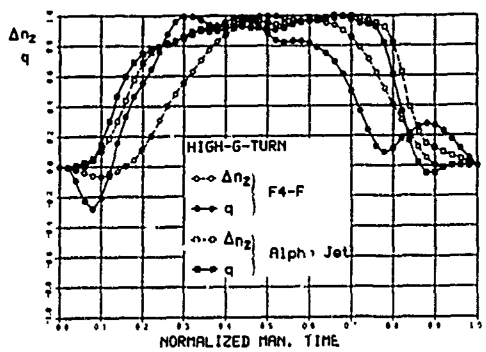


FIG.7.1 COMPARISON OF STANDARDIZED MANOEUVRES OF TWO AIRCRAFTS

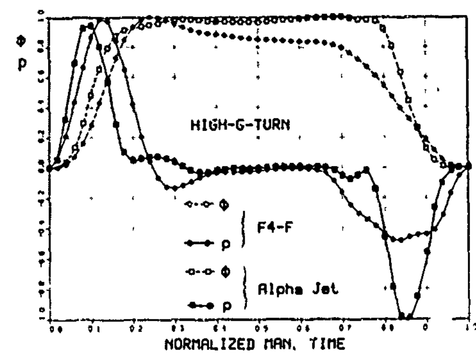


FIG.7.2 COMPARISON OF STANDARDIZED MANOEUVRES OF TWO AIRCRAFTS

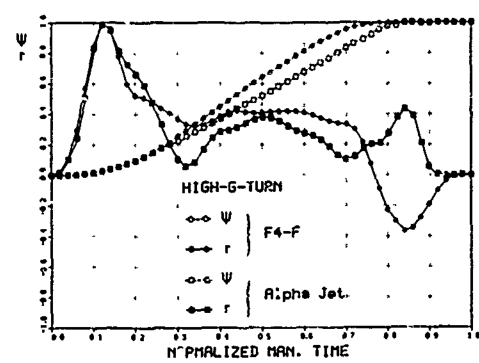


FIG.7.3 COMPARISON OF STANDARDIZED MANOEUVRES OF TWO AIRCRAFTS

3. MANOEUVRE MODEL

The procedure of the manoeuvre model is shown in Fig. 8 as a flow chart. As input, standardized parameters are used. First, the boundary conditions have to be determined, that means

- manoeuvre time, T_{Man}
- maximum load factors, n_z, n_y
- maximum bank angles, ϕ

Using the standardized parameters the transformation into real time is performed. In order to do the response calculation in the conventional manner, the control deflections are determined in the following simple manner:

- roll control ξ, by applying roll- and yaw equations
- pitch control η, using the steady pitch equation taking into account the symmetrical aileron deflection
- yaw control ζ, by applying sideslip- and yaw equations

The response calculation is done in real time, but for the purpose of checking the results with respect to the standardized manoeuvres, the response parameters are normalized.

In a comparison of the parameters between input and output of the manoeuvre model, the standardization is checked. In the case of conformity of main parameters of the response calculation with the standardized parameters, the output-parameters are considered to be verified. These verified data represent the model parameters for structural load calculation.

In Fig. 9.1 - 9.4 the comparison of normalized parameters for the High-G-Turn manoeuvre is shown.

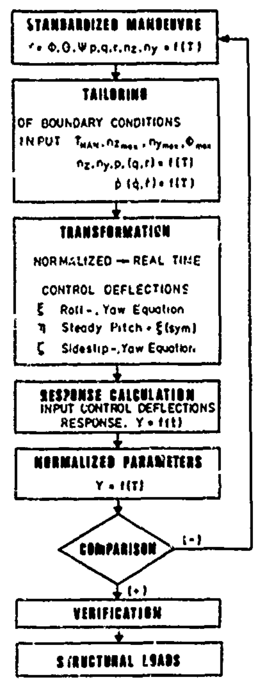


FIG. 8 MANOEUVRE MODEL

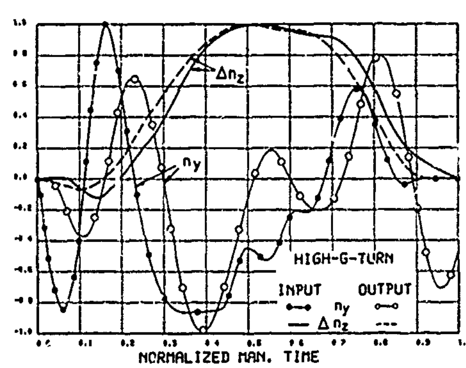


FIG. 9.1 COMPARISON OF NORMALIZED PARAMETERS, MANOEUVRE MODEL

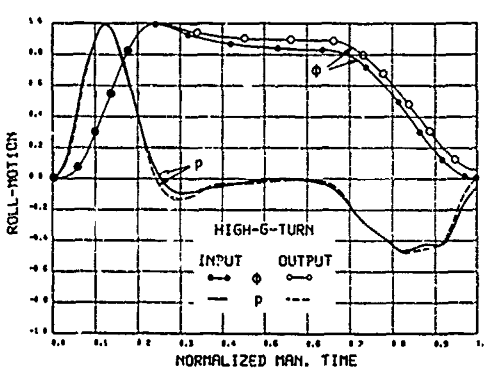


FIG. 9.2 COMPARISON OF NORMALIZED PARAMETERS, MANOEUVRE MODEL

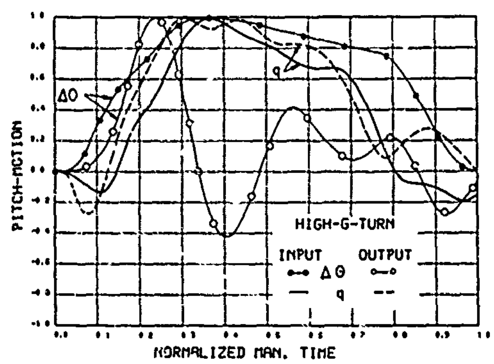


FIG.9.3 COMPARISON OF NORMALIZED PARAMETERS, MANOEUVRE MODEL

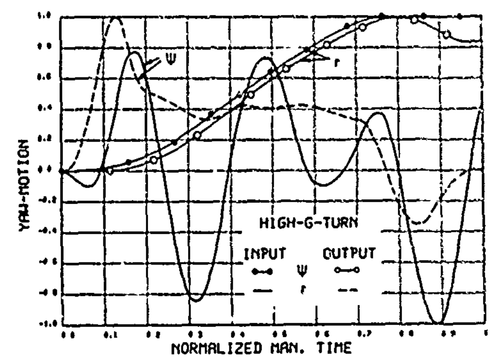


FIG.9.4 COMPARISON OF NORMALIZED PARAMETERS, MANOEUVRE MODEL

4. DETERMINATION OF EXTREME OPERATIONAL LOADS

The verified standardized parameters of the manoeuvre model are to be considered as a manoeuvre with mean parameters. For deriving the extreme manoeuvres, the main parameters of the manoeuvre model are scaled up to the extreme values to be obtained. The extreme values can be assumed with reference to design parameters required by specifications (MIL-Spec.) e.g.

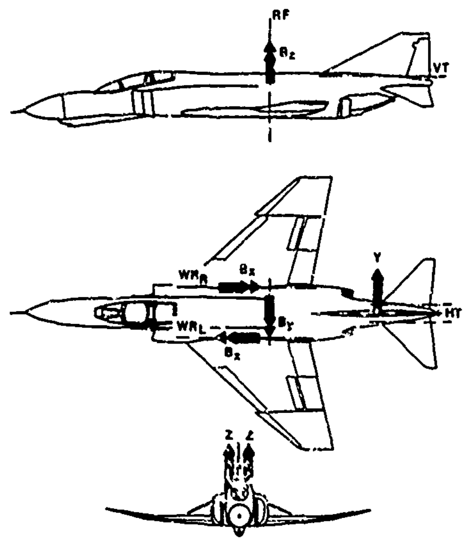
- vertical load factor for rolling pull out
- maximum roll control deflection attainable at the manoeuvre speed to be considered.

	$T_{MAN.}$ [s]		n_z		n_y		ϕ [°]	
	mean	extr.	mean	extr.	mean	extr.	mean	extr.
FULL AILERON REV	10	11	5.0	6.5	0.4	0.5	100	100
HIGH-G-BARREL ROLL O.T.	20	5.6	4.0	5.0	0.12	0.3	360	360
HIGH-G-BARREL ROLL UN	20	6.8	3.5	4.5	0.12	0.4	360	360
HIGH-G-TURN	8	5.3	5.0	6.5	0.25	0.5	90	90
ROLLING ENTRIES + PULL OUT	17	7.5	5.0	6.5	0.15	0.4	100	100

TABLE 1 MODEL PARAMETERS FOR LOAD ANALYSIS

Table 1 shows the mean values and the assumed corresponding extreme values for the manoeuvre time (T_{Man}), load factors (n_z, n_y), the angles of bank (ϕ).

For determination of the extreme parameters the maximum values of the mean parameters for the 5 analysed manoeuvres have been scaled up to the load factors required by MIL-8861 for rolling pull out. The determination of the extreme manoeuvres is performed by the same procedure as for the mean manoeuvres, but applying extreme boundary conditions (Fig. 8)



$B_x WR_R$	BENDING RIGHT WING ROOT
$B_x WR_L$	BENDING LEFT WING ROOT
B_y RF	BENDING LATERAL REAR FUSELAGE
B_z RF	BENDING VERTICAL REAR FUSELAGE
Z HT	SHEAR HORIZONTAL TAIL
Y VT	SHEAR VERTICAL TAIL

FIG.10 STATIONS FOR LOAD ANALYSIS

For the extreme manoeuvres the loads on the following main structural components have been analysed as shown in Fig. 10.

- bending right on wing root
- bending left on wing root
- bending vertical on rear fuselage
- bending lateral on rear fuselage
- shear on horizontal tail root
- shear on vertical tail root

For the High-G-Turn manoeuvre the extreme operational manoeuvre parameters are plotted in Fig. 11.1 - 11.4, the extreme operational loads in Fig. 11.5 - 11.7, and the control deflections in Fig. 11.8. The parameters and loads are plotted as normalized values. For the normalization the values are related to the maximum values indicated in the diagrams.

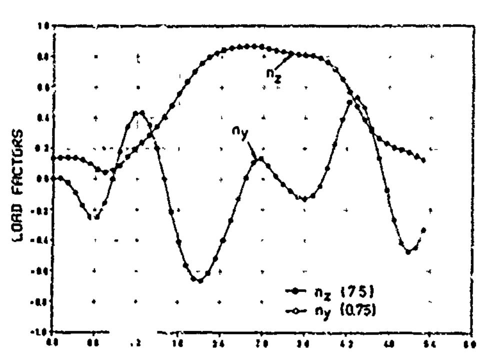


FIG.11.1 EXTREME OPERATIONAL LOADS AND PARAMETERS HIGH-G-TURN

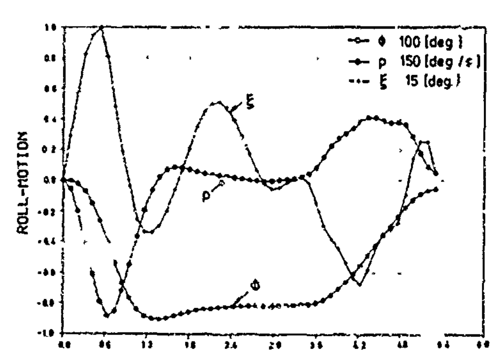


FIG.11.2 EXTREME OPERATIONAL LOADS AND PARAMETERS HIGH-G-TURN

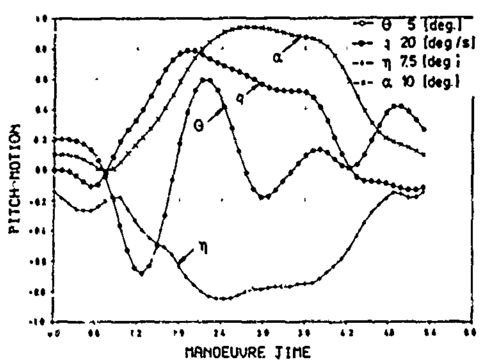


FIG.11.3 EXTREME OPERATIONAL LOADS AND PARAMETERS HIGH-G-TURN

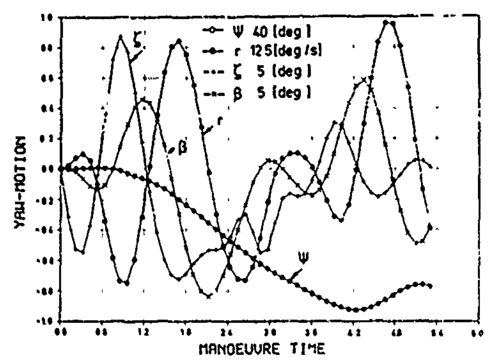


FIG.11.4 EXTREME OPERATIONAL LOADS AND PARAMETERS HIGH-G-TURN

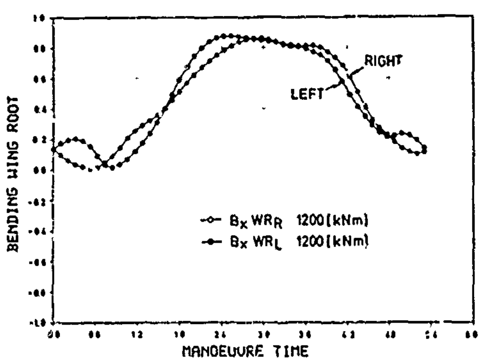


FIG.11.5 EXTREME OPERATIONAL LOADS AND PARAMETERS HIGH-G-TURN

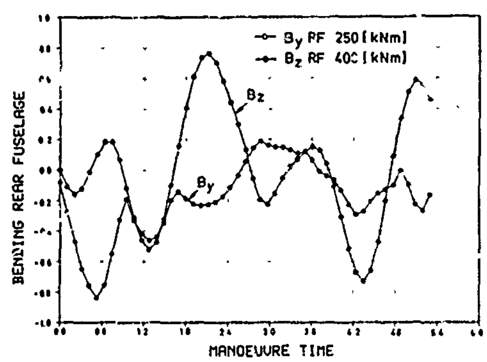


FIG.11.6 EXTREME OPERATIONAL LOADS AND PARAMETERS HIGH-G-TURN

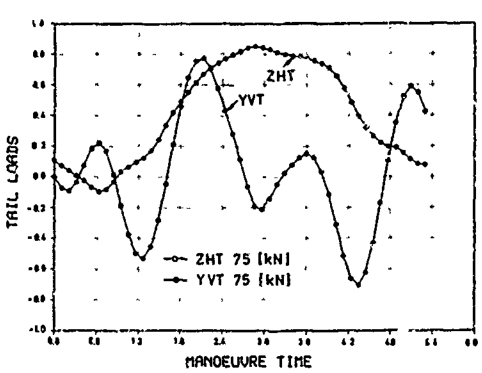


FIG.11.7 EXTREME OPERATIONAL LOADS AND PARAMETERS HIGH-G-TURN

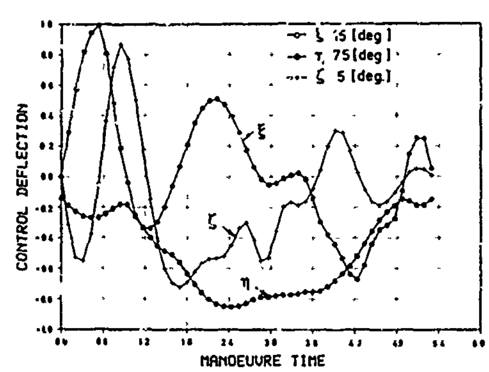


FIG.11.8 EXTREME OPERATIONAL LOADS AND PARAMETERS HIGH-G-TURN

Up to now the evaluation of operational manoeuvres has been performed for the following 5 manoeuvres:

- full aileron reversal
- high-g-barrel roll over the top
- high-g-barrel roll underneath
- high-g-turn
- rolling entries + pull out

The control deflections plotted in Fig. 12.1 - 12.3 show an interesting course for the five individual operational manoeuvres. In three manoeuvres alternating control deflections have been found, especially roll and yaw controls.

In detail:	Numbers of alternating deflections aileron	rudder
high-g-turn	4	4
full aileron reversal	3	3
rolling entries	2	2

The control deflection course in high-g-barrel rolls occurs in one direction only. For all manoeuvres the pitch control deflections show a moderate course.

Concerning the vertical load factor shown in Fig. 12.4, the course alternating the most is caused by the rolling entries and the full aileron reversals. In Fig. 13.1 - 13.5 the structural loads on main components versus manoeuvre time are plotted. Looking for correlations and alternations the following facts may be stated.

- the wing root bending correlates to the vertical load factor (Fig. 13.1 and 12.4)
- the lateral bending on the rear fuselage shows a simular time course as the load on the vertical tail (Fig. 13.3 - 13.5)
- the horizontal tail loads changing the most are found at rolling entries and full aileron reversal manoeuvres. During these manoeuvres two load peaks occur consecutively (Fig. 13.4)
- the vertical tail loads alternating the most are obtained at full aileron reversal and hig-g-turn manoeuvres (Fig. 13.5). For each of these manoeuvres at least four load peaks can be counted.

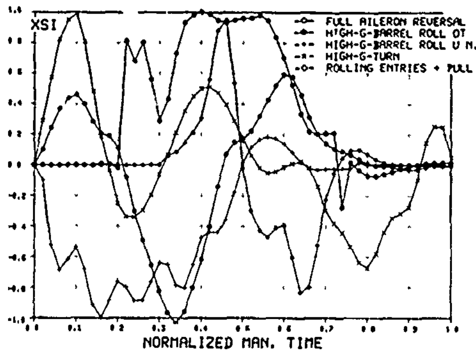


FIG.12.1 PARAMETERS
EXTREME OPERATIONAL MANOEUVRES

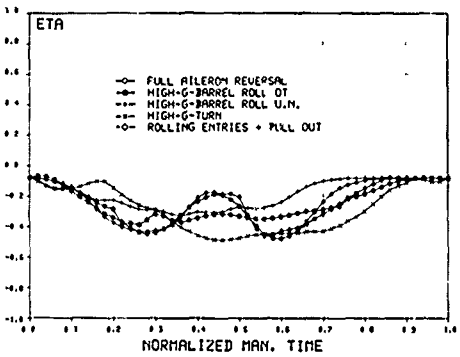


FIG.12.2 PARAMETERS
EXTREME OPERATIONAL MANOEUVRES

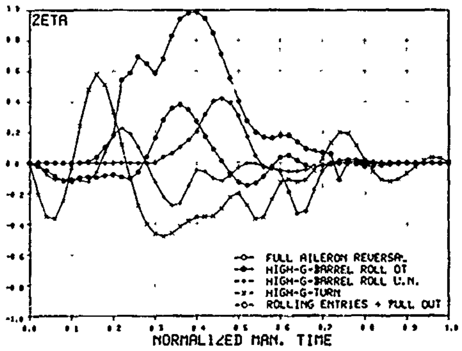


FIG.12.3 PARAMETERS
EXTREME OPERATIONAL MANOEUVRES

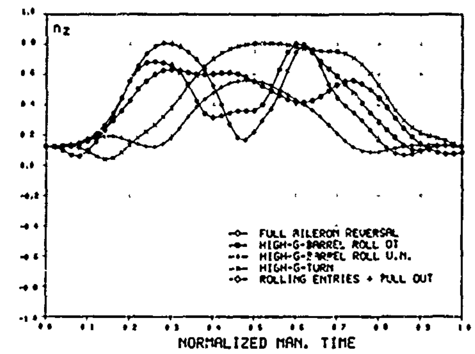


FIG.12.4 PARAMETERS
EXTREME OPERATIONAL MANOEUVRES

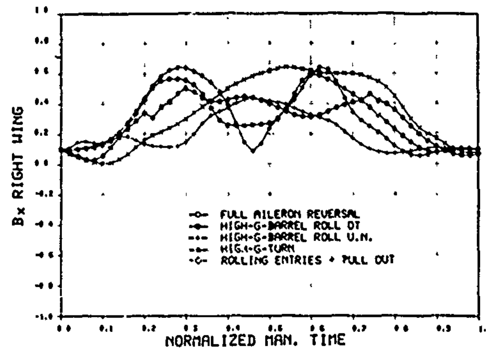


FIG.13.1 STRUCTURAL LOADS
EXTREME OPERATIONAL MANOEUVRES

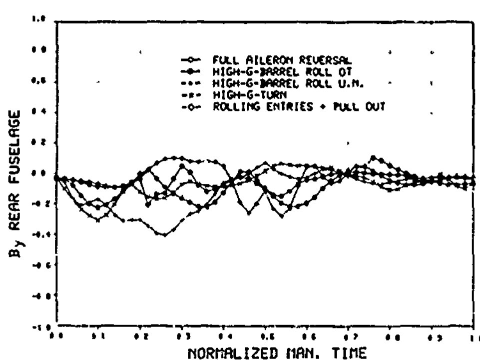


FIG.13.2 STRUCTURAL LOADS
EXTREME OPERATIONAL MANOEUVRES

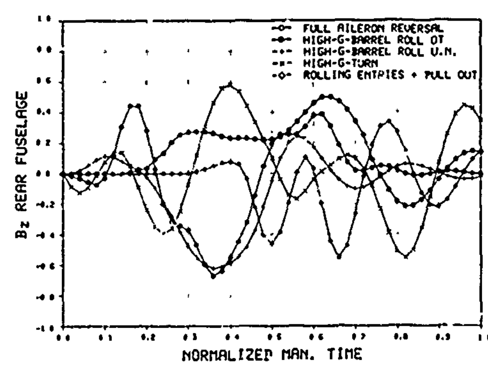


FIG.13.3 STRUCTURAL LOADS
EXTREME OPERATIONAL MANOEUVRES

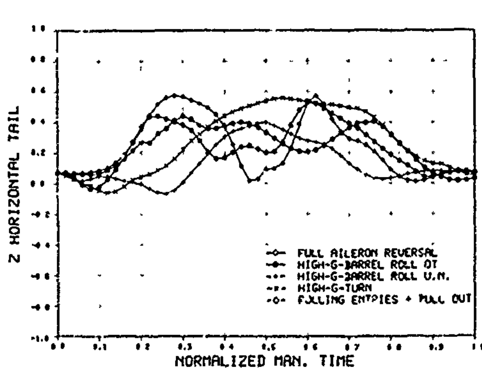


FIG.13.4 STRUCTURAL LOADS
EXTREME OPERATIONAL MANOEUVRES

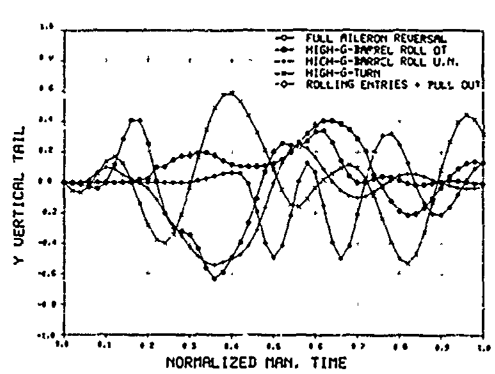


FIG.13.5 STRUCTURAL LOADS
EXTREME OPERATIONAL MANOEUVRES

5. COMPARISON OF EXTREME OPERATIONAL LOADS WITH DESIGN LOADS REQUIRED BY MIL-8861

In the design requirements [3] several flight conditions are specified, distinguishing between

- symmetrical flight conditions - pitching manoeuvres
- asymmetric flight conditions - yawing manoeuvres
  - rolling manoeuvres

For these manoeuvres, the displacements of the cockpit control are specified. Fig. 14 shows in a sketch the longitudinal, lateral, and directional control displacement time history. For comparison, the vertical load factor and the structural loads on the main components for all MIL-manoeuvres have been calculated. The results are plotted in the same manner as for the operational manoeuvres.

In Fig. 15, the load factors are presented. At a glance, a moderate course of the load factor during all manoeuvres is evident. Fig. 16.1 - 16.5 show the loads on the wing, rear fuselage and the tail planes where the load factors and the loads have been normalized with the design values, e.g. n_z (design) = 8.0 equalling 1.0

In table 2 the maximum values of the main load parameters, the structural loads for MIL-Manoeuvres, and the extreme operational manoeuvres are presented. The main parameters are absolute values, but the loads have been normalized by the design loads. This summary shows that the extreme operational structural loads are lower than the design loads specified by MIL-8861. The load level is about 66 % of the symmetrical pitch manoeuvres, respectively 60 % of the unsymmetrical rudder manoeuvre. But the frequency of control deflections and of structural loads for operational manoeuvres is higher than the frequency that results from design requirements.

	n_z max.	n_z min.	n_y	p [°/s]	β [°]	B_xWR	B_yRF	B_zRF	ZHT	YVT
ROLL 180°	0.90	-3.2	0.53	203	3.6	0.22	0.37	0.62	-0.38	0.59
ROLLING PULL OUT	6.50	+3.9	0.55	124	4.7	0.97	0.31	0.88	0.54	0.77
ROLL 360°	1.30	-1.1	0.28	210	1.8	0.34	0.39	0.35	-0.18	0.27
RUDDER KICK	1.10	+0.5	0.83	20	7.5	0.18	0.09	1.00	0.08	1.00
ABRUPT PITCHING ∧	8.0	+0.8	0	0	0	1.00	1.00	-	1.00	-
ABRUPT PITCHING ⁄‾\	8.0	+0.9	0	0	0	1.00	0.57	-	1.00	-
FULL AILERON REVERSAL	6.5	+0.5	0.49	123	4.1	0.66	0.22	0.63	0.53	0.61
HIGH-G-BARREL ROLL OT.	5.0	+0.6	0.25	177	2.0	0.51	0.21	0.48	0.44	0.40
HIGH-G-BARREL ROLL UN.	4.5	+0.7	0.40	164	2.7	0.46	0.40	0.59	0.40	0.52
HIGH-G-TURN	6.5	+0.3	0.50	132	4.2	0.65	0.30	0.54	0.56	0.56
ROLLING ENTRIES + PULL OUT	6.5	+0.5	0.40	139	1.9	0.64	0.27	0.52	0.57	0.48

TABLE 2 MAXIMUM VALUES OF MAIN LOAD PARAMETERS AND STRUCTURAL LOADS
MIL - MANOEUVRES / EXTREME OPERATIONAL MANOEUVRES

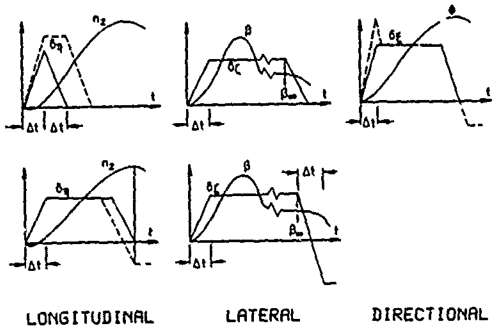


FIG.14 COCKPIT CONTROL DISPLACEMENT MIL-A-008861

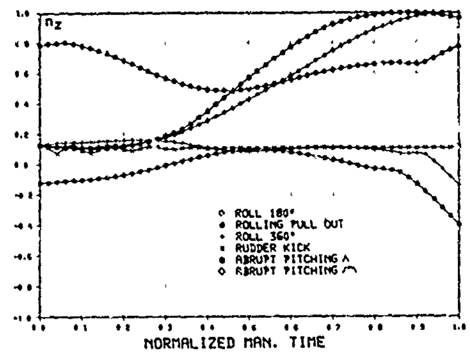


FIG.15 LOAD FACTORS - MIL - MANOEUVRES

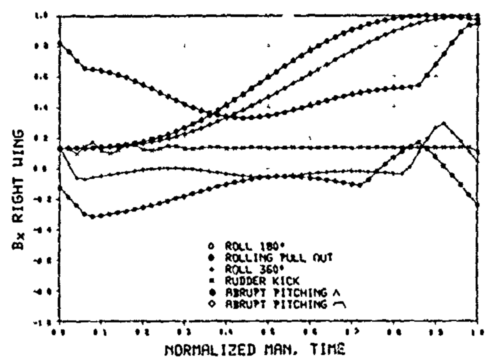


FIG.16.1 STRUCTURAL LOADS MIL-MANOEUVRES

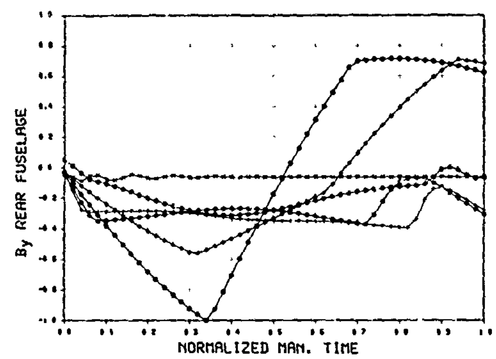


FIG.16.2 STRUCTURAL LOADS MIL-MANOEUVRES

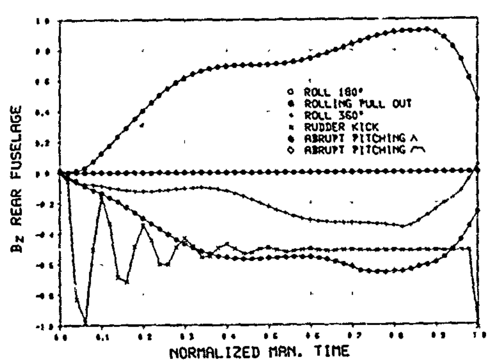


FIG.16.3 STRUCTURAL LOADS MIL-MANOEUVRES

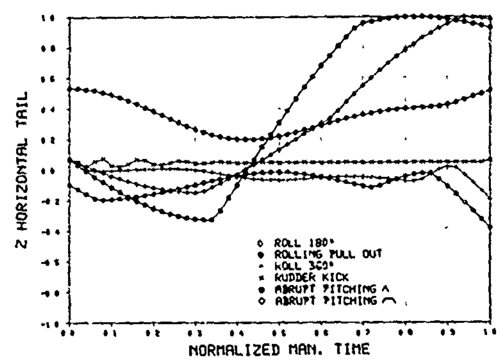


FIG.16.4 STRUCTURAL LOADS MIL-MANOEUVRES

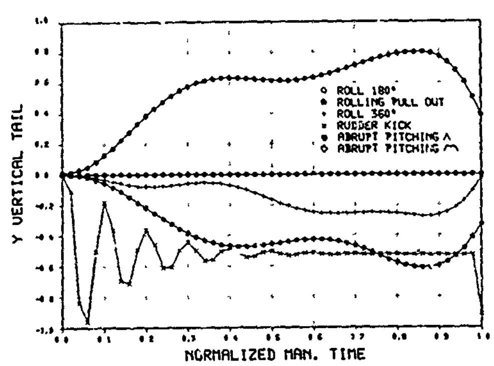


FIG.16.5 STRUCTURAL LOADS MIL-MANOEUVRES

6. POTENTIAL ASPECTS FOR FATIGUE DESIGN

Fatigue load prediction and monitoring is only as good as the knowledge of the magnitude and the frequency, namely the load parameters espected and monitored in service. The potentiality of the manoeuvre model allows the realization and the evaluation of long-time measurements of the relevant parameters. The recording should include all fatigue-relevant data, such as mass configuration (weight, C/G, external stores) and the data describing the flight profiles (speed, altitude, flap setting). For standardized manoeuvres, the manoeuvre model makes available

- the time history of the main parameters and the loads on the main structural components
- the correlation of the main parameters and the loads

The spectra of relevant parameters for several operational manoeuvres can be determined by systematic measurements made in service. Applying the manoeuvre model and the parameter spectra, the resultant load spectra for the expected mission of an aircraft can be established. That means the manoeuvre model can be applied for fatigue load prediction and fatigue monitoring as well.

7. CONCLUSION

For the manoeuvres evaluated a standardization of relevant parameters of motion is feasible, and the results can be made compatible with the equations of motions by smoothing and tuning. It could be shown that the standardization is in agreement for the evaluated operational manoeuvres flown by a second aircraft type. The parameters of the standardized manoeuvres are used in a manoeuvre model for the determination of the control deflections.

In the manoeuvre model, the mean values or the extreme values of parameters and the structural loads can be ascertained. For five operational manoeuvres, extreme structural loads on main components are presented and discussed. A comparison of the extreme operational loads evaluated with the design loads required by MIL-8861 indicates moderate load sequences but higher load levels for MIL-manoeuvres.

The present state of evaluation has led to the following results:

- the manoeuvres evaluated can be considered as Standard Manoeuvres in normalized time and amplitude for parameters of motion
- the time history of relevant parameters including control deflections and thus loads occuring during operational manoeuvres, can be determined
- the relationship of operational parameters and loads acting on individual primary structural components has been verified
- the extreme operational loads are determinable by applying main load parameters as specified in the regulations e.g. n_z(max), $\dot{P}$(max) etc. or by extreme value distributions measured in service

In conclusion the statement can be made that the measurement of few relevant parameters of motion is sufficient for standardized manoeuvres to derive design loads for static and fatigue design. The evaluation might be improved by:

- availability of a greater number of in-flight manoeuvres measured for each manoeuvre type flown by several aircraft including those with active controls
- evaluation of other manoeuvre types, if possible all Standard NATO Manoeuvres applying the manoeuvre model
- systematic recording of relevant parameters in service will permit specific load evaluation for static and fatigue design for various manoeuvres and/or missions.

This is a first step towards determination of loads from operational data in service while keeping expences at a justifiable level. The results obtained so far encourage us to continue the investigations, with improvement of recording and evaluating accuracy being clearly possible.

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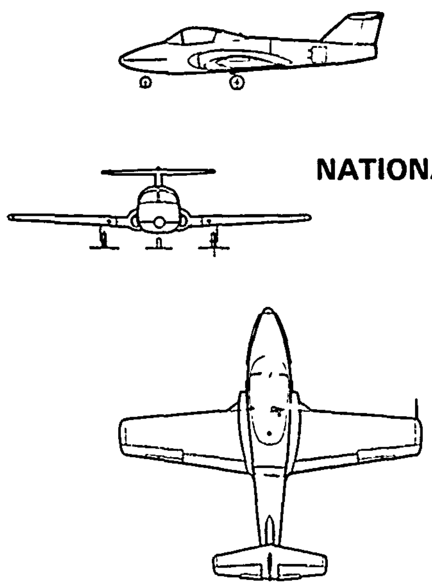
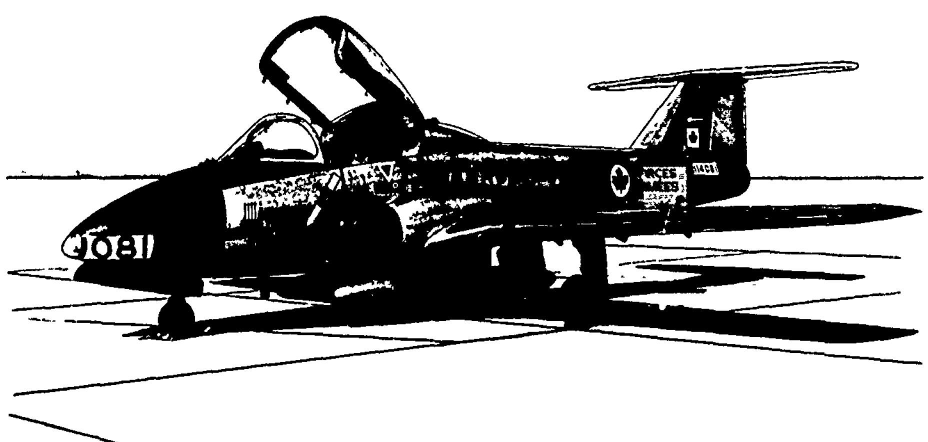
TUTOR (CL-41A)
TAIL FLIGHT LOAD SURVEY

BY

A. PAQUIN, STRESS ENGINEER
CANADAIR LIMITED
P.O. BOX 6087
MONTRÉAL, QUÉBEC H3C 3G9
CANADA

J. SKOTNICKI, DAS ENG 6-3-4
NATIONAL DEFENCE HEADQUARTERS
OTTAWA, ONTARIO K1A 0K2
CANADA





TUTOR TAIL FLIGHT LOAD SURVEY

SUMMARY

An in-flight structural failure of the Canadair Tutor aircraft horizontal tail fitting resulted in the initiation of a joint Canadian Forces/Canadair flight load survey. An instrumented test aircraft was flown to the extremes of its structural envelope, and strains were recorded at 43 different locations on the rear fuselage and the empennage of the aircraft.

This report describes the test instrumentation used, the calibrations performed on the test article, and the resulting formulation of load equations for estimating shear, bending moment and torque at various locations on the rear fuselage, the vertical and the horizontal stabilizer. The various missions and manoeuvres flown to gather the necessary data are discussed, as well as the data acquisition, verification and reduction methods.

1.0 INTRODUCTION

On the evening of May 3rd, 1978 a Canadair Tutor (CL-41A) crashed while engaging in the aerobatic performance as a part of the air show in Grand Prairie, Alberta, Canada. Specifically, the crash occurred while the aircraft was executing a manoeuvre called the Level Triple Roll at an altitude of 300 feet above ground level. The pilot died in the accident.

The subsequent accident investigation identified the cause factor to be the failure of the horizontal stabilizer attachment fitting in the Tee tail of the aircraft (see Fig.1). The failure resulted from the fatigue crack originating at the radius (point A in Fig. 1), which had weakened the fitting to an extent where complete separation occurred during the application of the manoeuvering load.

This distressing event raised suspicions in the minds of many that the aeroplane may be structurally deficient to safely withstand its current usage, and that a more thorough investigation of the operational loads, as well as of the airframe strength may be necessary. Thus an ambitious program was born, tailored specifically to evaluate the flight load environment of the aircraft, to be followed by the fatigue and damage tolerance analyses of the critical areas in the structure.

To confirm the requirement for such a program a preliminary flight load survey was conducted by the Canadian Forces Aerospace Engineering Test Establishment (AETE) in Cold Lake, Alberta (Ref. 1). This survey, which took approximately 10 weeks to complete, had two objectives to fulfill. One was to identify those flight manoeuvres which caused high asymmetric loads in the horizontal tail, and the other was to measure the relative magnitude of these loads for both the aerobatic display role and the training role. Both objectives were satisfactorily met and the results indicated that certain manoeuvres did indeed produce high rolling moments, the maximum recorded being 110% of the design limit, and was achieved in the four-point roll.

The Canadian Forces (CF) operate a sizeable fleet of Tutor aircraft in the training role. Also, the same aircraft type is used in the air display role. About 45% of the training fleet aircraft is equipped with counting accelerometers and recorders for the purpose of collecting "G" exceedance data. All of the Snowbird Aerobatic Display Team aircraft are equipped with the same system. To date the computer data bank contains approximately 107000 hours of data collected over the years (Ref. 2). This data, which is continuously updated, is used to periodically calculate the remaining fatigue life of the Tutor wing root, hitherto considered to be the most critical area in the airframe.

2.0 PROGRAM OBJECTIVES

The accident at Grand Prairie indicated that the wing root may not be the most critical area. There could be other areas, more severely stressed under certain loading conditions, that needed investigating. Accordingly, the decision was made to concentrate the program on the rear fuselage and the empennage of the airframe.

The program objectives were: - (Ref. 3)

a) To confirm by means of a flight load survey the maximum design limit loads used in the structural analysis of the CL-41A aft fuselage and empennage structure.

b) To establish for the various mission profiles of the CL-41A the frequency of application and magnitude of the loads applied to the aft fuselage and empennage structure.

The detail objectives of the load survey was to:-

a) Determine loading conditions which produced the critical structural loads.

b) Determine and define suspected new critical loading conditions.

c) Determine the frequency and magnitude of the loads for the various manoeuvre profiles of the CL-41A.

d) Verify the structural loads used in the airframe design.

Essentially the program, which became a joint CF-Canadair venture, was broadly divided into three phases:

a. Phase One. The design of the instrumentation and application of the strain gauges to the test article including the calibration; a responsibility of Canadair Ltd.;

b. Phase Two. Preparation of the test program, actual test flying and data acquisition; a responsibility of CF; and

c. Phase Three. Data reduction and subsequent fatigue and damage tolerance analyses; a responsibility of Canadair Ltd.

This paper presents the various steps performed in order to reach the stated objectives.

3.0 INSTRUMENTATION

Canadair identified 108 different locations for possible strain monitoring. Each of these locations received two strain gauge bridges, one primary (or main), and one secondary (or back-up). Two different types of strain gauges were used depending on the kind of loading anticipated in a given location: the end load type for tension, compression and bending such as in sparcaps and flanges, and the shear load type for shear and torsion in webs, skins and torque tubes (Ref. 4). The bridges were installed at the zones shown in Fig. 2. An example of bridge location is shown on Fig. 3.

Flight Parameter Recorders

The right hand seat in the cockpit of the test aircraft had been removed in order to install the Standard Aircraft Instrumentation package (SAIS). The SAIS consisted of the following main components (see Fig. 4).

a. Signal Conditioning Units (SCUs)
b. Power Distribution Centre (PDC)
c. Pulse Code Modulation (PCM) Encoder
d. Control Panel
e. Precision Power Supply (PPS); and
f. Bell and Howell MARS 2000 Tape Recorder.

The general layout of the Tutor SAIS is shown in Fig. 4. The wiring harness extended to all areas of the aircraft where sensors were mounted. The signal conditioning Printed Circuit Boards (PCBs) were designed and produced by AETE's Flight Test Instrumentation Section. Power to the SAIS installation was provided through the two static inverters and a Power Distribution Centre and was controlled independently of other aircraft systems through the instrumentation package control panel. The PPS package provided a stable and precise power for transducer excitation and to the SCUs. The Programmable Data Acquisition System (pDAS) was a Pulse Code Modulation (PCM) encoder capable of encoding 128 parallel signals into a serial PCM digital signal for on-board recording and telemetry transmission to a ground receiving station. The pDAS was programmed to record 74 measurements in addition to time pulse at a rate of 66.5 samples/sec with 11 bits per word for a resolution of 0 to 2047 (Ref. 5).

In addition to the above instrumentation, a separate Flight Test Monitoring Unit (FTMU) was designed and manufactured by AETE specifically for this program. The purpose of the FTMU was to allow project engineers to monitor the missions by providing to them instantaneous information via telemetry on flight parameters and the resulting loads acting on the rear fuselage and the empennage of the test aircraft. Such data reduced the risk of dangerous manoeuvres and minimized flying mission time by instantaneously indicating to the engineers whether or not it is safe to proceed to the next test point, without having to land the aircraft to analyse data before proceeding. The information on specification and design details of FTMU can be found in Ref. 5. The test aircraft was also equipped with a nose boom, which provided static and dynamic pressure, angle of attack and side slip data.

4.0 LOAD CALIBRATION

Canadair Calibration

In order to determine the correlation between the bridge readings and the external load, a load calibration was carried out (Ref. 6). This calibration consisted of applying load at various locations listed in Table 1.

The load was applied using one or two manually operated hydraulic jacks. Each jack was fitted with a pre-calibrated load cell.

For each loading point the maximum load to be applied was previously determined to avoid any structural damage. In order to account for the hysteresis of the structure and/or the gauges, an "exercise" run was done before the actual calibration. This "exercising" consisted of applying the maximum load on the loading point and coming back to zero. The calibration loads were applied in 31 increments covering 3 full cycles. Each calibration increment represented 20% of the maximum load applied. The bridges and load cells outputs were recorded at each of the loading increments. To ensure that the operations were progressing normally, an output hand-check was performed on selected bridges.

The recording equipment consisted of:

1 Bruel and Kjaer Strain Indicator (Type 1526).
1 Bruel and Kjaer Multipoint Selector and Control (Type 1544).
10 Bruel and Kjaer Multipoint Selectors (Type 1545).

This equipment read the load cells and bridges output and directly converted it into indicated strains. The data was then checked for overflows and stored on a Hewlett-Packard HP-9825A cassette.

Each cassette contained 6 load cases or 198 files. This was then transferred, with a Hewlett-Packard HP-1000 computer, onto a tape compatible with IBM 370 tape drives which was used for data reduction.

AETE Calibration

The AETE ground calibration of the strain gauges was accomplished by applying loads to the designated surface contact points using the load jig and wiffle tree arrangement. The calibration loads amounted to 55% of the anticipated flight loads. The calibration procedure revealed several insensitive gauges which had to be replaced. This in turn necessitated recalculation of the original load equation matrix, which now became Version Two. Early into the flying phase, within the first three flights, it became apparent that certain strain gauges were quite sensitive to the temperature and carry-over effects. These sensitivities manifested themselves through changes in gauge zero points on the ground immediately following engine starts, and through erratic trends in shear and bending moment values obtained in flight. Consequently, test flying had to be interrupted to conduct another temperature survey of the gauges, but this time over a wider range than was done previously.

5.0 BRIDGE SELECTION

The flight recording system limited the number of bridges to be read to 43.

In order to select the more accurate bridges, the following criteria were used:

Bridge Response and Linearity

The bridges were selected according to their response to the various inputs. Gauges were chosen which had linear and adequate response and those gauges which showed non-linear or negligible response were rejected. This assumed that the gauge in question was expected to respond. For example, the outboard gauges on the horizontal stabilizer were unlikely to respond to applied fin loads but were not rejected unless they also failed to respond to loads applied in their region of the horizontal stabilizer.

Bridge Duplication

In some areas, bridges were only duplicating others and were kept as replacements in the eventuality of a malfunction.

Temperature Sensitivity

Even though the bridges were temperature self-compensating, some bridges were affected by temperature variation. In order to detect this, two

temperature surveys were carried out and only the bridges with minimal temperature changes were selected.

Offset Limitation

In order to avoid difficult signal conditioning and noisy results, the offsets were limited to 15 mV.

Carry-Over Effect

Some bridges on the horizontal tail were affected by a loading applied on the opposite side of the tail. These bridges were not used.

Of the 216 initially installed bridges, 10 were rejected by Canadair in the initial temperature survey for oversensitivity, 11 were found not functioning under load calibration, and another 17 had to be rejected on the grounds of overly critical offset effects. Of the remaining 178 gauges, 43 were surveyed over a wider temperature range (20 C degrees) resulting in the substitution of another 12. Analysis of the calibration data and the results obtained in the first three test flights before flying was terminated, indicated that significant carry-over effects were still present. Several procedures were considered to eliminate this undesirable phenomenon, such as in-flight temperature soak and roller coaster flight technique to establish a test zero condition before and after each manoeuvre. All were judged unsuitable and discarded due to the extra time requirement in a mission conflicting with the endurance of the test aircraft. The difficulty was solved by the use of a computer program which permited the determination of temperature sensitivity and the carry over effect of each bridge. This resulted in a greatly improved matrix of load equations (Version Three).

6.0 LOAD EQUATIONS

The load equations were developed based on the calibration of strain-gage installations on aircraft structures. The simplest relation between the output of a strain-gage bridge and the loads (shear, moment, and torque) was expressed by the following linear equation (Ref. 7):

$$V = C_1 R_1 + C_2 R_2 + \dots\dots + C_i R_i + \dots\dots + C_m R_m = \sum_{i=1}^{m} C_i R_i$$

Where V is the load to be estimated (Shear or Moment)
R_i is the reading of bridge #i.
and C_i is a constant for bridge #i.

A computer program had been developed to evaluate the bridge constants and to resolve the above equation based on the theory defined in Reference 7. This computer program also computed the probable error of each load equation so that the irrelevant bridges and redundancy could be eliminated. This allowed the achievement of more accurate and reliable load equations.

The following load equations were derived for the rear fuselage and empennage of Tutor aircraft:

a) Horizontal Stabilizer (Figure 5)

The load equations for Shear (F_z), Bending moment (M_x) and Torque (M_y) at Station 7 on the port and starboard, and at stations 30 and 54 on the port side were produced.

As example, the total shear in Z-direction is calculated using the following equation (Port side, station 7):

$$F_z = 0.18843\ R_{21P} + 0.65534\ R_{22P} + 0.048491\ R_{31P} + 0.38388\ R_{66S} - 0.14457\ R_{41P}$$

b) Vertical Stabilizer (Figure 6)

The load equations for Shear (F_y), Bending Moment (M_x) and Torque (M_z) at Station 15 were produced:

As example, the Bending Moment Equation is:

$$M_x = 3.7029\ R_{70P} + 3.9516\ R_{72P} + 21.194\ R_{77S} - 17.861\ R_{78S} + 8.9268\ R_{81P} - 7.6252\ R_{82P}$$

c) Rear Fuselage (Figure 7)

The load equations for Shears (F_y & F_z) and Moments (M_x, M_y & M_z) at fuselage Station 370 were produced.

As example, the Torque Equation is:

$$M_x = -2.0039\ R_{9S} - 58.550\ R_{11P} - 60.123\ R_{12S}$$
$$-1.2554\ R_{15P} + 8.0060\ R_{16P}$$

7.0 DATA ACQUISITION

All data generated during the flight test phase was recorded on board the test aircraft, and later suitably reformatted for reduction by Canadair. In addition, selected data was transmitted through telemetry to the Flight Test Monitoring Unit (FTMU) located on the ground, for simultaneous real time processing. As mentioned before, this was very useful to the project engineers controlling a mission from the Flight Test Control Room (FTCR). It proved invaluable from the flight safety point of view on several occasions, when any of the load parameters approached dangerously close to previously estimated limit load, or even exceeded it. On such occasions the test pilot was immediately advised to terminate the test flight and return to base for consultation. Each exceedance was carefully analyzed and the exceeded limit load extrapolated to the "G" limit of the aircraft. The new design limit load was thus established and the test point reflown if necessary to confirm the analysis. The higher design limit was allowed only when the Margin of Safety in the corresponding critical area was positive. This approach was used on several occasions to extend the limit load and allow the aircraft to fly a test point which was on the limit of the "G" envelope of the aircraft.

Event Marking System

An event marking system was designed into the data acquisition system which enabled individual manoeuvres to be located on any data tape.

Flight Testing

The flight load survey program consisted of operating an aircraft, with an instrumented and calibrated aft fuselage and empennage, within the limit structural design envelope to measure the resulting loads. The test flights included the extremes of the envelope.

Flight testing was grouped into two main objectives: Military Specification Manoeuvres and Mission Profiles. The objectives were to establish through tests what loads actually existed in the aft fuselage and the empennage of the aircraft during the limit conditions encountered in flight. All test flights were flown during daytime under visual meteorological conditions. All MIL SPEC manoeuvres were performed at the Primrose Lake Evaluation Range (PLER), a Canadian Forces testing facility in northern Alberta. The missions were continuously monitored by the Project Engineers manning the FTMU. All missions were flown in clean configuration at the maximum forward centre of gravity, except for one aft CG mission. The Snowbird formation profiles were flown out of the Canadian Forces Base Moose Jaw, the team's home base. The actual test flying commenced on 27th November, 1979 and was finally completed on 20th May, 1982. The program required 75 sorties and consumed 130 flying hours including all the necessary calibration, test and ferry flights.

Military Specification Manoeuvres

All the basic manoeuvre requirements of MIL SPEC. MIL-A-8871A applicable to the military trainer category were flown. These manoeuvres included symmetrical pull-ups and push-downs with normal, abrupt, and abrupt with checking inputs, rolls and rolling pull-outs, side slips, rudder kicks, deceleration device extensions which were all performed at the forward C of G. The manoeuvres of pull-ups, rudder kicks, and deceleration device extensions were performed at the aft C of G of the aircraft. Other manoeuvres were spins with and without external tanks, spins with speed brakes extended, and inverted spins. Refer to Tables 2 and 3 for more specific information.

Mission Profiles

The mission profiles (Ref 8) flown in this program could be divided into three major categories: those flown in the training role, the solo aerobatic role, and the formation aerobatic role. The training role missions were flown in accordance with the Canadian Forces training guidelines and consisted of the following profiles: Clear Hood, Formation, Cross-Country, Navigation, Instrument, and Maintenance Test Flights (Ref. 8). The two latter ones duplicated actual missions flown by the Canadian Forces Snowbird Demonstration Team during their air shows. Also, to make them as realistic as possible, the formation profiles were flown by the Snowbird team members using the test aeroplane in the two most severe formation positions, the Second Line Astern and the Outer Right Wing. (Fig 8)

8.0 DATA REDUCTION & ANALYSIS

8.1 Data Verification

On receipt of flight load data tapes, a computer program verified the following items:

Time

The information about starting and finishing time for any manoeuvre or event provided by the pilot was compared with the "Event Mark" (Ref. Section 7) registered on the tape. This check gave assurance that the information stored on tape corresponded to the specific manoeuvre or event expected.

Saturation

As mentioned in Section 3, the "PDAS" limit values were 0 and 2047. When data reached one of these values, a plot of "PDAS value vs time" was done (Fig. 9 & 10). From this plot, it was possible to determine if the saturation was caused by:

1) Input Error (Spikes as in Fig. 9)

In that case the corresponding data was rejected.

2) High Loads (Smooth curve as in Fig. 10)

That saturation indicated that the maximum expected output had been exceeded. The actual value could be extrapolated or the flight had to be repeated with new limits for this bridge.

8.2 Data Reduction

Manoeuvre Flights

These flights were used to determine the maximum load to be used in the static analysis.

For each flight, the maximum load registered for each location (Ref. Section 6) was compared with the load used for the static analysis. When the recorded load was higher, a new margin of safety was calculated, as explained in Section 7.

Mission Flights

The mission flights were used to produce the load spectra for the various mission profiles of the CL-41A. The flight data from AETE missions (Section 7) were reduced event by event using a Rainflow and Peak Between Zero (PBZ) counting methods (Ref. 9 and 10) to produce a load spectrum for each event.

These spectra were then factored to suit the current CF usage and combined to produce a total load spectrum for an aircraft mission. A further combination was applied to give a load spectrum for each of the following aircraft roles:

a. Trainer
b. Snowbird Formation (aerobatic)
c. Snowbird Solo (aerobatic)

The PBZ spectra were used for graphical comparison of the three roles as shown in Fig. 11.

The Rainflow spectra, being more reliable, are used in the Fatigue and Damage Tolerance Analysis. (Ref. 11 and 12)

9.0 RESULTS AND DISCUSSION

Almost from the very beginning of flight testing it became evident that certain MIL SPEC manoeuvres would generate loads approaching or exceeding the estimated safe load limits, in some cases substantially below the stated "G" limits of the airframe. All exceedances were analytically investigated, and where considered prudent, the safe load limits were extended to allow for the repeat of the test point. In all instances except one the Margin of Safety was greater than zero. The one exception was the vertical fin bending moment which indicated a Margin of Safety of -0.06 based on a conservative analysis. This occurred in the solo aerobatic profile, as a result of one manoeuvre, the abrupt 4-point roll, a manoeuvre which has since been discontinued. After considerable thought it was decided to increase the safe limit based on the fact that a

static strength test on one of the prototypes demonstrated adequate strength in that area without failure, generating a Margin of Safety of 0.19.

In general the flight test phase of the program was considered extremely successful. It indicated in many instances that the loads imposed on the airframe at the extremities of the load factor (G) envelope were higher than expected by the manufacturer or the operator. The vast amount of data collected during the course of flight testing phase was turned over to Canadair for their use in the Fatigue and Damage Tolerance Analysis.

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TABLE 1

CALIBRATION DATA (Ref 3)

COMPONENT	CALIBRATION LOAD POSITIONS
Aft Fuselage	Vertical shear, side shear, and torque applied at tail.
Horizontal Stabilizer	Vertical loads to produce shear, bending moment and torque.
Vertical Stabilizer	Side loads. Also a moment load applied by means of the horizontal stabilizer.
Elevator and Control Rods	At all hinge locations on the fixed surface and all control rods.
Rudder and Control Rods	At all hinge locations on the fixed surface and all control rods.
Dive Brake	On the actuator support structure and dive brake surface.
Miscellaneous	Horizontal stabilizer forward attachment link.

TABLE 2

BASIC MILITARY SPECIFICATION MANOEUVRES (Ref 5)
(MIL-A-8871A)

MIL SPEC REFERENCE A-8871A (Para)	DESCRIPTION OF TEST	WEIGHT (lb)	CG	ALT (ft)	FLAP (deg)	EAS (kt)	LOAD FACTOR (g)	MAX* PILOT EFFORT (lb)	TEST SUMMARY
4.5.1	Normal Symmetrical Pull-Up	6,900	Fwd	5,000 5,000 14,000	0	420 297 **326	7.33	As Req	Critical positive wing loads
4.5.2	Normal Symmetrical Push-Down	6,900	Fwd	5,000 5,000	0	425 435	-3.0 -1.0	As Req	Critical negative wing load.
4.5.9	Abrupt Co-ordinated Rolling Pull-out	6,900	Fwd	5,000	0	300	5.87	50	Combined horizontal & vertical tail loads
4.5.3	Normal Uncoordinated Rolling Pull-out	6,900	Fwd	5,000 5,000	0	**420 435	5.87 -1.0	60 60	Critical wing torsion
Ref 3-2	Abrupt Uncoordinated 360° Roll with Abrupt Pull-up	6,900	Fwd	5,000	0	360	5.87	As Req	As Defined by AETE
4.5.6	Abrupt Symmetrical Pull-Up	6,900	Fwd	5,000	0	**420	7.33	As Req	Critical down load on horizontal tail
4.5.7	Abrupt Symmetrical Pull-up with Abrupt Checking	6,900	Fwd & Aft	5,000	0	**420 **297	7.33	As Req	Critical up and down loads on horizontal tail
4.5.8	Abrupt Symmetrical Push-Down with Abrupt Checking	6,900	Fwd	5,000	0	435	-1.0	As Req	Critical torque on horizontal tail
4.5.14	Landing Approach Pull-up	7,000	Fwd	5,000	40	140	4.0	As Req	Gear-down landing approach configuration
4.5.16	Rudder Manoeuvre-High Speed Steady Sideslip - Rudder Reversed	6,900	Fwd	5,000	0	250 350 425	1.0	180	Critical vertical tail loads
4.5.17	Rudder Kick Manoeuvre with Abrupt Return	6,900	Aft	5,000	0	**435	1.0	180	Critical vertical tail loads
4.5.18	Rudder Kick Manoeuvres for Landing Approach	7,000	Fwd	5,000	40	140	1.0	300	Critical vertical tail loads
4.5.5	Steady State Yaw Manoeuvres	6,900	Fwd	5,000 5,000 5,000	0	**435 350 250	1.0	300	Critical vertical tail loads
4.5.10	Abrupt Uncoordinated Rolling Pull-out with Abrupt Checking	6,900	Fwd	5,000	0	**420	5.87	60	Critical combined horizontal and vertical tail loads
4.5.11	Abrupt Coordinated 180 Degree Roll	6,900	Fwd	5,000	0	**420	5.87	60	
4.5.12	Abrupt Uncoordinated 180 Degree Roll	5,900	Fwd	5,000	0	**426	5.87	60	
4.5.13	Abrupt Uncoordinated 360 Degree Roll	6,900	Fwd	5,000	0	**429	5.87	60	
4.5.20	Deceleration Device Extension	6,900	Fwd & Aft	5,000		**420 435	7.33 -1.0	As Req	Critical airbrake extension cases

* Maximum (MIL SPEC permissible) pilot effort where specified is as follows:
60 lb - aileron control column for maximum displacement
180 lb - rudder control force for displacement and return
300 lb - rudder control force for maximum displacement

** These points are the estimated buffet onset speed, at the load factors specified.

TABLE 3

SPINS-MIL SPEC REQUIREMENTS (Ref 5)

DESCRIPTION OF TEST	WEIGHT (lb)	CG	ALT (ft)	FLAP (deg)	DIVE BRAKES	LOAD FACTOR (g)	TEST SUMMARY
Tail Loads in Spinning Manoeuvres (Smooth Upright Spins)	Opt	Fwd	25,000	0	Retr	-	Normal five turn spins carried out right, and left, with aileron and entry at 1.0, 2,5 and 3.5 g.
Tail Loads in Spinning Manoeuvres	Opt	Fwd	15,000	0	Retr	-	Normal five turn spins carried out right and left with neutral, pro-spin, and anti-spin ailerons from entry at 1.0, 2.5, and 3.5 g.
Tail Loads in Spinning with Dive Brakes Fully Extended	Opt	Fwd	25,000	0	Ext	-	With dive brakes fully extended right and left spins of five turns each.
Tail Loads in Inverted Spins	Opt	Fwd	25,000	0	Retr	-	As defined by AETE
Tail Loads in Spinning with External Tanks	Opt	Fwd	25,000	0	Retr	-	As defined by AETE

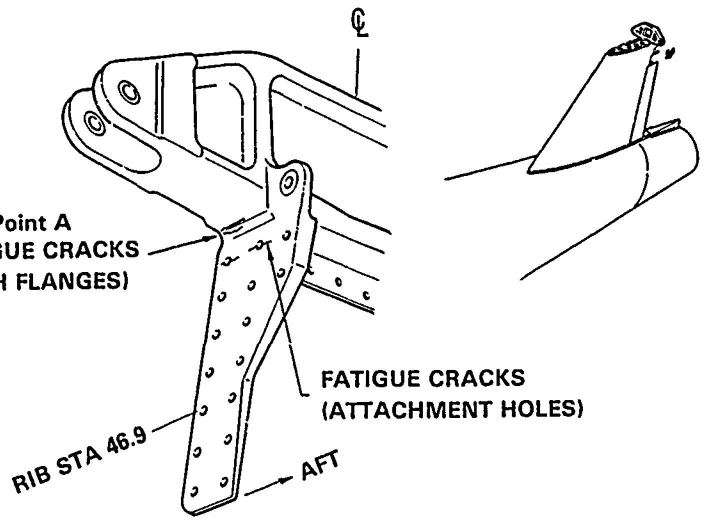


Figure 1 - Horizontal Stabilizer Rear Attachment Fitting

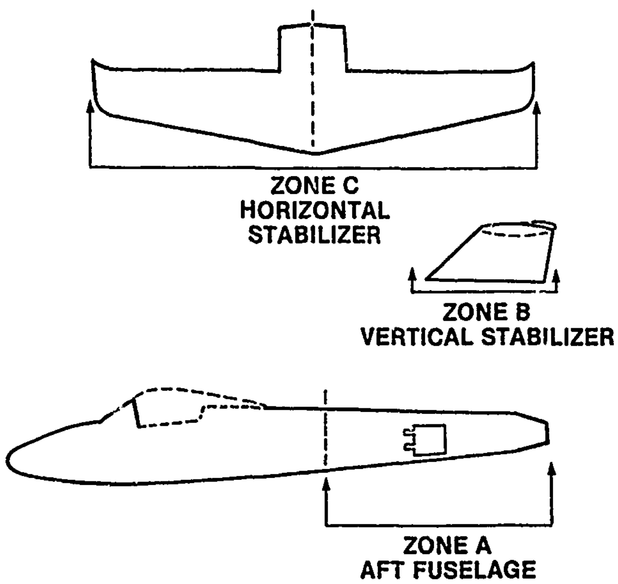


Figure 2 — Zones for Bridges

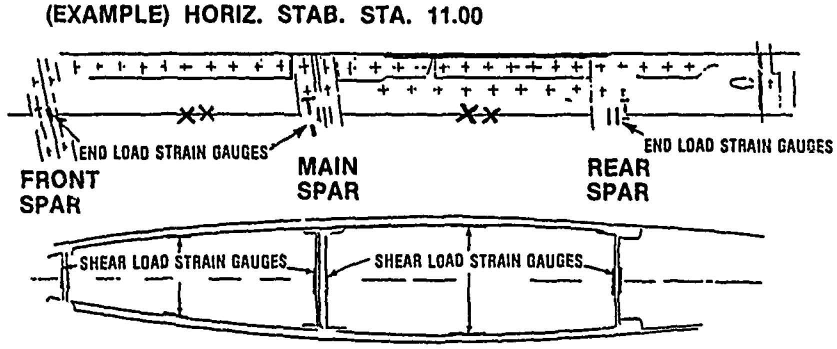


Figure 3 - Typical Strain Gauge Location

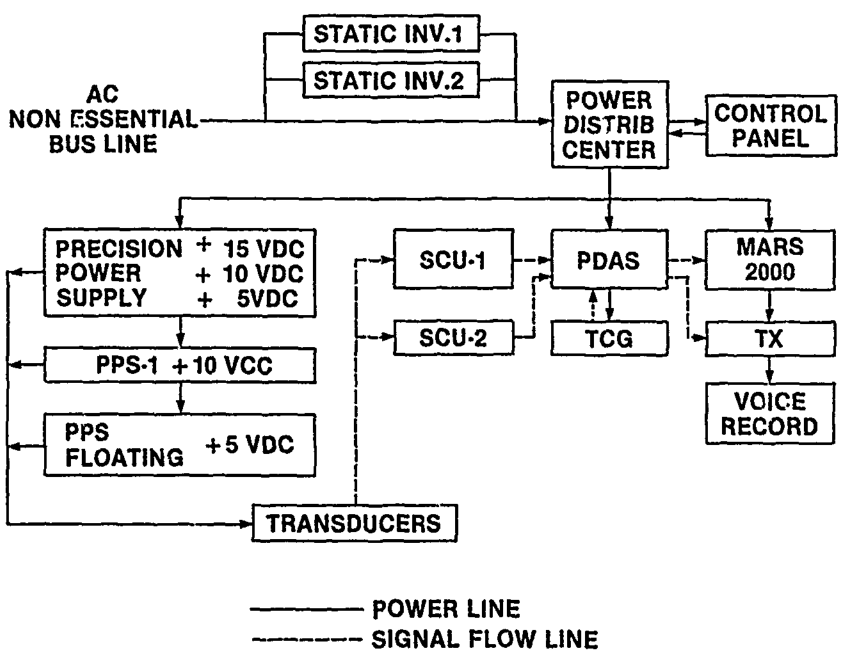


Figure 4 - General Layout - CL-41A SAIS

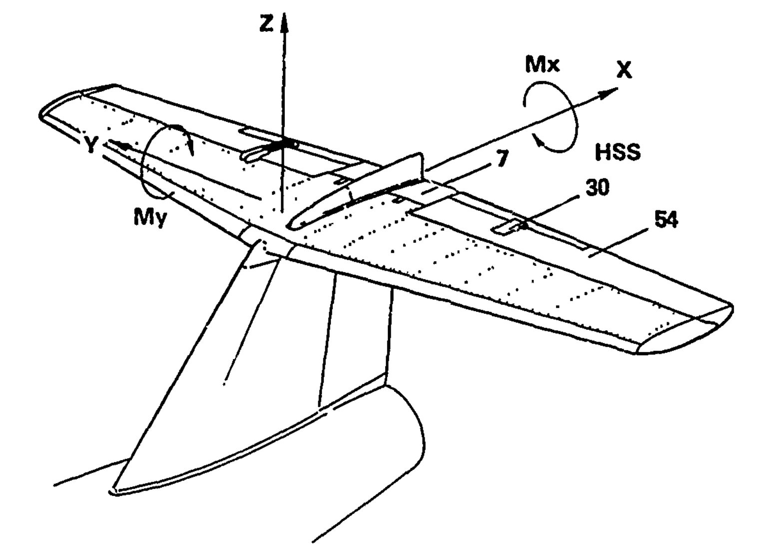


Figure 5 - Horizontal Stabilizer

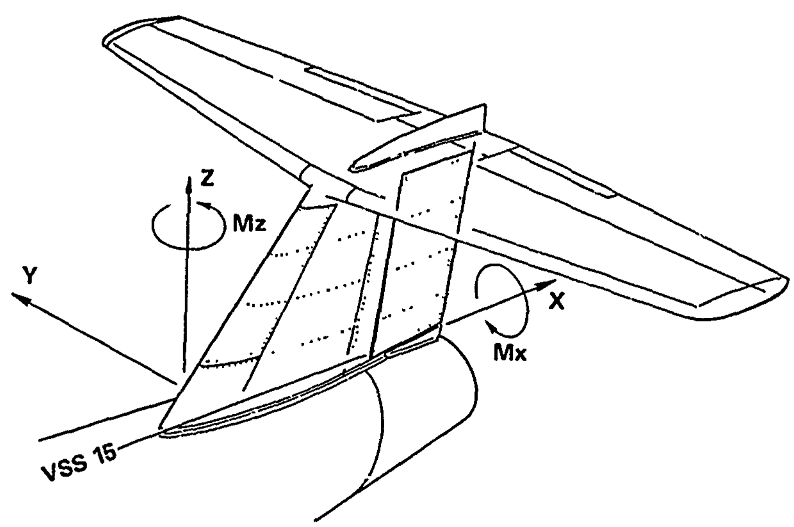


Figure 6 - Vertical Stabilizer

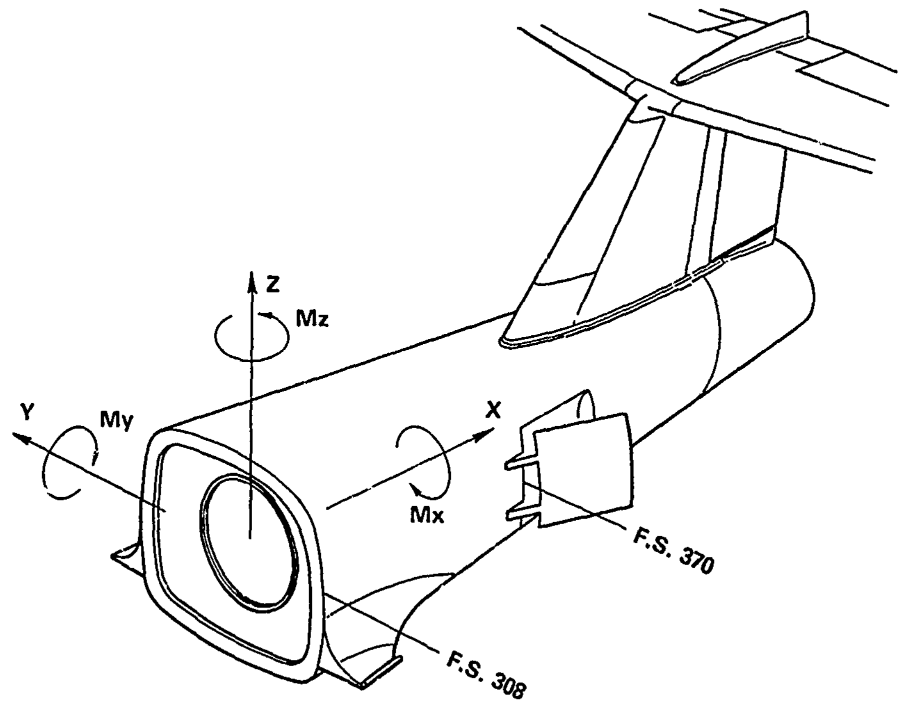


Figure 7 - Rear Fuselage

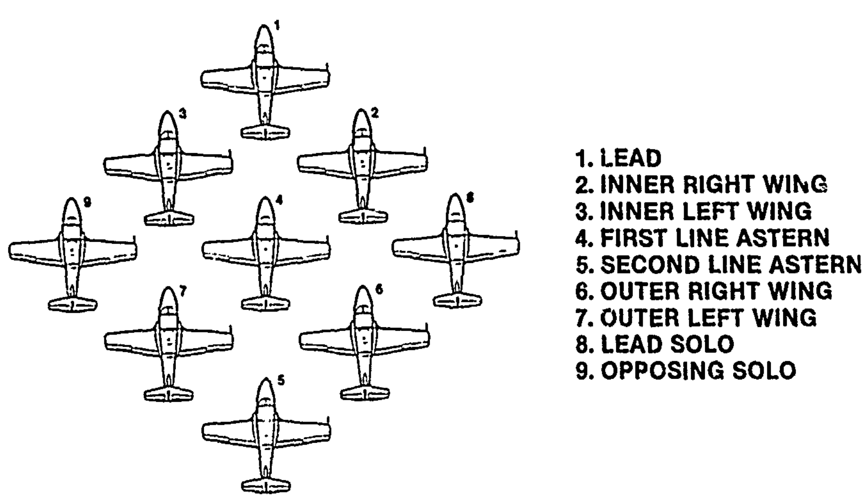


Figure 8 — Snowbird (Aerobatic) Formation Positions

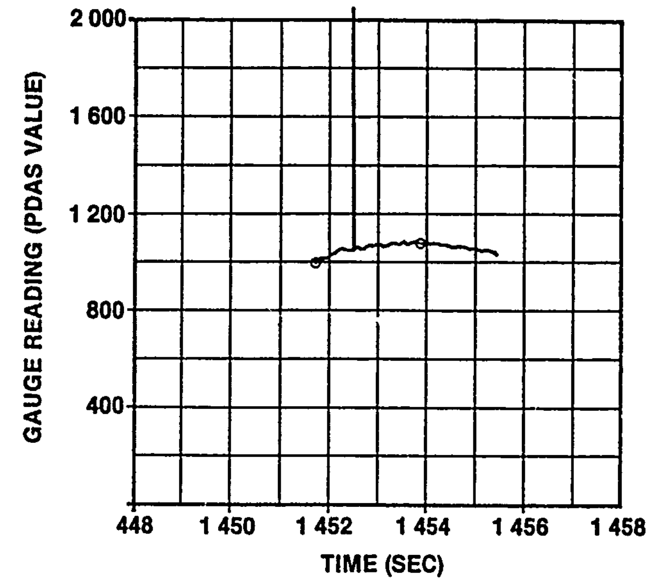


Figure 9 — Gauge Reading Spike Example

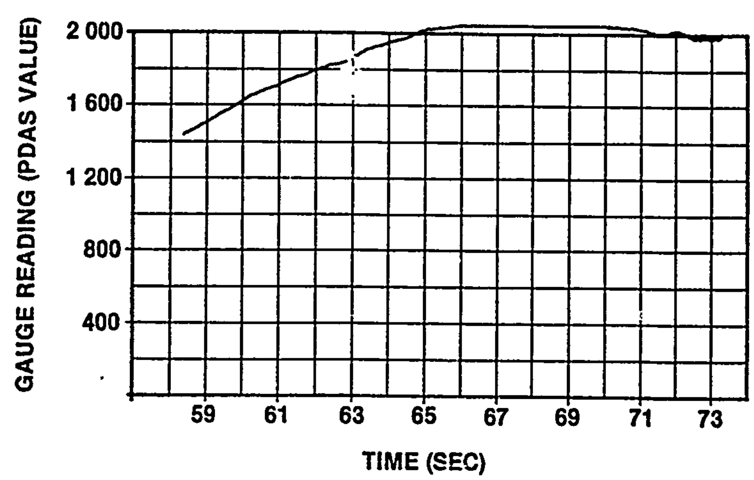


Figure 10 — Gauge Reading Saturation Example

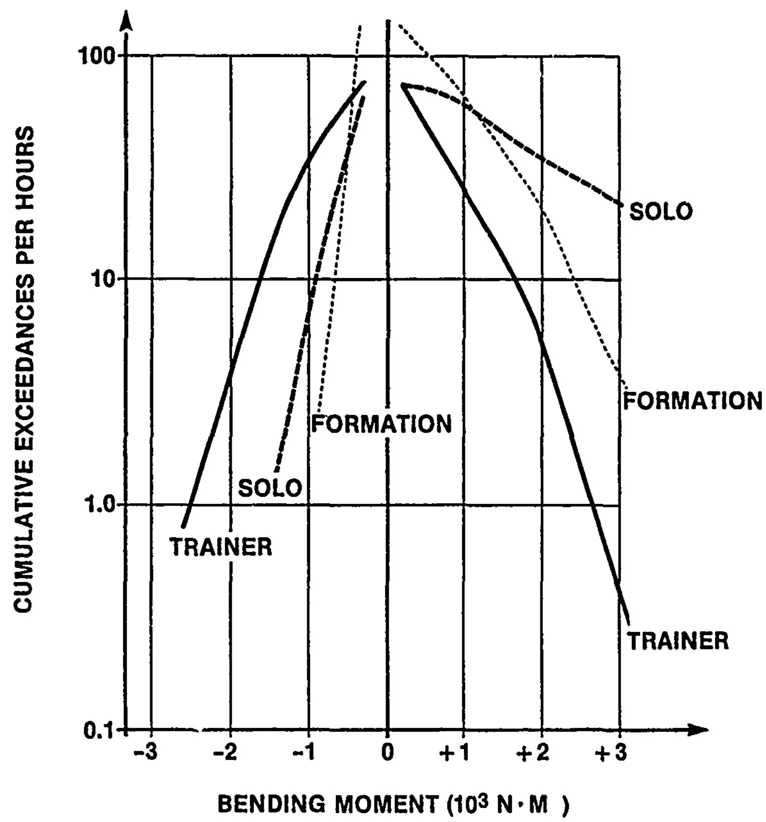


Figure 11 — Load Spectra Example Port Horiz Stab. Peak Between Zero

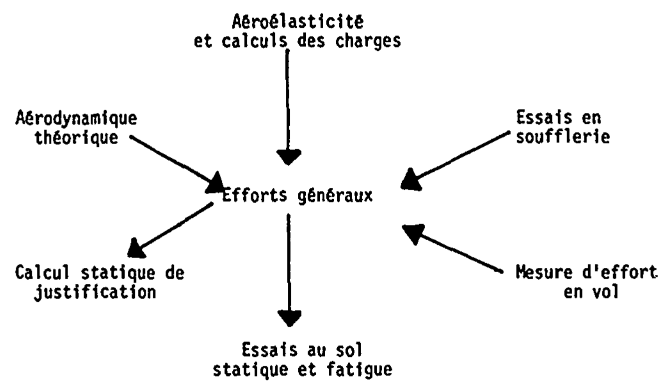
ANALYSE AEROELASTIQUE ET IDENTIFICATION DES CHARGES EN VOL

C. PETIAU et M. de LAVIGNE

AVIONS MARCEL DASSAULT-BREGUET AVIATION
78, quai Carnot
92214 - SAINT-CLOUD

1 - INTRODUCTION

Le calcul de charges est classiquement fondé sur une modélisation élastique par la théorie des poutres longues. Cette hypothèse, impliquant une correspondance biunivoque entre les courbes d'efforts généraux (effort tranchant, moment de flexion, moment de torsion) et les répartitions de contraintes dans la structure, il en résulte une certaine facilité pour les échanges de données entre les diverses disciplines intervenant dans l'analyse des structures, qui se résument aux seuls efforts généraux, comme nous l'illustrons ici :



Ceci conduit pour les mesures en vol à instrumenter des sections de voilure ou de fuselage avec des jauges de contraintes qui permettent théoriquement de restituer les efforts généraux de la section par combinaison linéaire de leurs signaux ; l'opérateur de combinaisons est établi à partir des réponses des jauges sur des chargements d'étalonnage au sol ; une méthode de moindre carré est utilisée dans les cas de mesures redondantes.

Le modèle élastique de la structure n'intervient pas dans cette procédure.

Les chargements des cellules d'essais statique ou de fatigue qui sont eux-mêmes construits à partir de courbes d'efforts généraux peuvent être aisément adaptés aux résultats mesurés en vol.

Cette technique simple est malheureusement contestable si on examine des avions dont la voilure est à faible allongement, ou si on veut obtenir une certaine précision dans les zones d'emplanture ; les contraintes n'étant plus alors déterminées seulement par les efforts généraux.

Pour l'analyse aéroélastique statique et le calcul des charges, nous avons été amenés à développer une procédure de couplage où les charges de pression issues des calculs d'aérodynamique théorique sont transférées directement sur le modèle éléments finis.

Cette méthode sophistiquée a l'inconvénient de compliquer singulièrement le problème du recalage des modèles sur les résultats de soufflerie ou de vol ; la première difficulté à dominer est que dans l'ancienne technique un chargement est représenté globalement sur l'avion par un vecteur facilement maniable à quelques dizaines de composantes alors que, sans précaution, il en comporte quelques milliers, voire dizaines de milliers sur un schéma d'éléments finis (voir planche 1).

C'est dans ce contexte que nous avons étudié des processus d'identification visant à répercuter les résultats des mesures en vol aisément dans nos modèles aéroélastiques complexes, et revoir, en conséquence, le plus rapidement possible les calculs de justification de la structure et les chargements d'essais statiques au sol.

Nous présentons des tests de cette méthode sur le Mirage 2000.

2 - CHARGES ET AEROELASTICITE STATIQUE AVEC ELEMENTS FINIS

Pour la compréhension du processus d'identification, nous rappelons les grandes lignes de notre technique de couplage aéroélastique directe sur éléments finis ; cette méthode qui fournit les modèles à recaler, est programmée dans la branche "CHARGE" de notre logiciel général ELFINI (voir référence 1).

Nous avons essayé de répondre à trois impératifs apparemment contradictoires :

- nécessité d'une précision de l'estimation des charges d'une qualité équivalente à celle du calcul des contraintes par éléments finis,
- sophistication devant rester homogène avec les hypothèses simples des calculs d'aérodynamique linéarisée, recalés sur des essais en soufflerie ou en vol,
- délais de préparation des données et coût ordinateur raisonnable.

Les opérations, dont le détail est donné en annexe, se déroulent en deux phases :

1 - Calcul volumineux mené indépendamment du couplage aéroélastique de la configuration massique et surtout des hypothèses de recalage empirique ; ce sont :

- la résolution du modèle éléments finis pour quelques centaines de chargements de base dans lesquels on décompose a priori tout chargement ; les chargements de base sont classés en cas unitaire de pression, d'inertie et d'effort concentré (ex : efforts train),
- les calculs des champs de pression aérodynamique par des méthodes de singularité, pour les mouvements de corps solide, les braquages de gouverne et des déformées de base des surfaces portantes.

De ces calculs, on extrait des opérateurs "concentrés" assez maniables.

- décomposition des champs de pression aérodynamique unitaires dans les chargements de base,
- lissage des déplacements, sous les cas de charges de base, dans les déformées de base des surfaces portantes,
- efforts généraux et contraintes en quelques centaines de points sensibles pour chaque chargement de base.

2 - A partir de ces opérateurs, on obtient par des calculs de faible volume :

- les coefficients aérodynamiques avion souple,
- le mouvement de l'avion en manoeuvre par intégration des équations de la mécanique du vol,
- l'évolution correspondante des efforts généraux et des contraintes aux points contrôlés,
- la recherche automatique des cas enveloppes dimensionnants,
- la reconstitution dans ces cas pour des analyses complètes de l'ensemble des déplacements du modèle E.F., ou des forces pour le transfert sur un modèle raffiné.

La définition exacte des configurations massiques et les recalages empiriques sur la soufflerie ou le vol interviennent au début de cette deuxième phase.

Les chargements des essais statiques correspondant aux cas de charges dimensionnants du modèle peuvent être générés automatiquement, à partir de la position des vérins de chargement ; on procède par une méthode d'optimisation quadratique visant à minimiser l'énergie élastique de la différence entre le cas de charge du modèle et celui reconstitué par les vérins, tout en respectant, dans la fourchette d'une précision donnée, les contraintes sur les points sensibles de la structure.

3 - IDENTIFICATION

3.1 - Instrumentation des essais en vol

Outre la saisie des paramètres généraux de la mécanique du vol et des braquages de gouvernes, l'installation d'essais est basée sur des jauges de contraintes ou des extensomètres ; leurs dispositions résultent d'un compromis entre la surveillance des points sensibles de la structure et la couverture des divers "chemins" d'efforts internes, un ordre de grandeur de 100 à 300 capteurs est raisonnable.

Nous présentons planche 2 l'installation de la voilure du Mirage 2000-01.

3.2 - Validation du modèle élastique par étalonnage au sol

La base de la méthode étant que les incertitudes ne proviennent que de l'aérodynamique, il faut qualifier de façon sûre l'opérateur linéaire , issu du modèle éléments finis, qui représente le passage entre les forces appliquées sur la structure et les réponses des jauges aux points de mesure en vol ; les erreurs à ce niveau étant pratiquement irrattrapables.

La procédure est la suivante :

- on mesure au sol les réponses des jauges sous divers chargements simples, par exemple, obtenus en chargeant l'avion en des points discrets (voir planche 3),
- on correlle pour chaque voie de mesure les résultats d'essais à ceux du calcul par éléments finis correspondant, en traçant la courbe déformation calculée fonction de déformation mesurée, pour les différents cas de charges ; théoriquement les points obtenus devraient s'aligner sur la première bissectrice ; on s'estime satisfait si on trouve une droite dont on prend la pente comme facteur de correction du modèle éléments finis (voir planche 4) ; on corrige ainsi aussi bien les défauts locaux du modèle éléments finis que les erreurs de gain éventuelles de l'installation d'essais.

On ne valide ainsi que la bonne résolution de l'hyperstaticité par le modèle E.F. ; d'expérience les rigidités, qui interviennent au deuxième ordre par l'aérodistorsion dans le calcul des charges

sont suffisamment bien prévues ; toutefois, s'il est nécessaire, nous disposons d'une méthode de recalage automatique du modèle éléments finis sur les essais de vibration (voir réf. 2).

3.3 - Manoeuvres et mesures en vol

On choisit des manoeuvres permettant de séparer le plus aisément possible chaque effet aérodynamique élémentaire (incidence, braquage gouvernes, etc.) ; il nous est apparu que les plus efficaces étaient des manoeuvres d'oscillation de tangage ou de roulis à fréquences variables (0,5 à 2 Hz) que nous appelons "stimulus", voir planche 4.

Ces manoeuvres durent quelques dizaines de secondes, le Mach et le badin doivent rester le plus constant possible ; l'amplitude du mouvement doit être limitée à quelques degrés pour rester dans une plage linéarisable.

Sur le Mirage 2000 prototype, les stimulus sont générés automatiquement par un boitier ajouté dans la chaine des commandes de vol électriques.

Pour cerner les non-linéarités d'incidence, on complète les stimulus par des montées en facteur de charge stabilisées.

Pendant ces manoeuvres, on doit acquérir les paramètres du mouvement de l'avion et les braquages de gouvernes à une cadence suffisante pour pouvoir les interpoller, moyennant quoi les mesures de jauges peuvent être sous échantillonnées (8 à 16 pt/sec. pour les stimulus).

3.4 - Prélissage des mesures en vol

Cette phase a pour but d'extraire des mesures en vol des effets aérodynamiques linéaires unitaires.

Soit qr le vecteur des paramètres aérodynamiques "rigides". En symétrique : incidence, vitesse de tangage, braquage des gouvernes.

On identifie l'opérateur supposé linéaire $[\partial\sigma/\partial qr]$ de réponse des jauges à chaque composante de qr par une méthode de moindres carrés, soit :

$$1 \qquad \sum_{\text{instants mesures}} [\sigma_{mes.} - (\partial\sigma/\partial qr)\, qr(tmes.)]^2 \quad \text{minimum}$$

Si la voie de mesures considérée est bien sensible à chaque composante de qr dans la manoeuvre considérée, les résultats sont bons (voir planche 5) ; dans le cas contraire, il faut prendre les composantes mal observables dans une autre manoeuvre plus propice, et considérer comme connus les termes correspondants de l'équation 1 ; en particulier, on procède ainsi pour les petites composantes symétriques des stimulus de roulis qui ne sont jamais parfaitement anti-symétriques.

Appliquée aux capteurs de mécanique du vol, une technique semblable (référence 3) permet d'identifier directement des coefficients aérodynamiques linéarisés avion souple qui seront dits "mesurés".

3.5 - Paramètres des champs aérodynamiques

Le modèle élastique ayant été qualifié à partir des étalonnages, on admet que les erreurs du modèle aéroélastique ne proviennent que des calculs d'aérodynamique.

On pose que les champs de pression théoriques de chaque effet aérodynamique rigide (incidence, braquage de gouverne en symétrique) sont modulés par des fonctions d'affinité inconnues dépendant linéairement de paramètre λi ; en choisissant comme fonction de recalage les interpolations d'une grille d'éléments finis, on donne aux coefficients λi inconnues la valeur des facteurs de correction des champs de pression aux noeuds de la grille (voir planche 2) ; soit pour les champs de pression réels des effets rigides, la relation :

$$\left[\frac{\partial Kp(M)}{\partial qrj}\right]\text{réel} = \left[\frac{\partial Kp(M)}{\partial qrj}\right]\text{théorique} \times \left(1 + \Sigma\lambda ij\phi i(M)\right)$$

Nous n'avons pas pris de paramètre spécifique pour les champs de pression des effets d'aérodistorsion (discrétisés par un vecteur qs) car leurs influences se séparent difficilement de celles des termes rigides en quasi-stationnaire ; nous utilisons 2 tactiques au choix :

1) ne pas recaler ces champs d'effet "souple" ; ce qui peut avoir quelque inconvénient quand ils coïncident en partie avec des effets rigides (exemple, débraquage d'une gouverne dû à la souplesse de la timonerie de commande),

2) corriger les effets "souples" en fonction des effets rigides en posant :

$$\left[\frac{\partial Kp(M)}{\partial qsj'}\right]\text{réel} = \left[\frac{\partial Kp(M)}{\partial qsj'}\right]\text{théorique} + \left(\left[\frac{\partial Kp(M)}{\partial qrj}\right]\text{réel} - \left[\frac{\partial Kp(M)}{\partial qrj}\right]\text{théorique}\right)[\alpha_{jj'}]$$

L'opérateur $\alpha_{j,j'}$ représente le lissage de chaque effet souple dans les effets rigides.

Nous détaillons en annexe comment par notre organisation de calcul, on obtient simplement la valeur et la dérivée en fonction des paramètres $\lambda_{i,j}$:

- des coefficients aérodynamiques avion souple $[C]$,
- des contraintes unitaires $[\partial\sigma/\partial qr]$ de chaque effet rigide, aérodistorsion éliminée.

Dans le cas du recalage des effets aérodynamiques "souples" en fonction des effets rigides, ces opérateurs ne sont pas linéaires en fonction des λ .

3.6 - Première technique d'identification : méthode de moindres carrés

On minimise par moindres carrés l'écart entre les contraintes unitaires et coefficients aérodynamiques calculés et ceux issus du prélissage des mesures, soit :

$$z(\lambda) = \sum_{\text{voies de mesure}} \pi \left[\frac{\partial\sigma}{\partial qr}\text{ mes} - \frac{\partial\sigma}{\partial qr}(\lambda)\right]^2 \text{ minimum}$$

La procédure de minimisation de la fonction de "coût" $z(\lambda)$ est détaillée en annexe ; elle est conduite par une méthode de Newton Raphson modifiée dans le cas non linéaire (recalage des effets souples).

Dans la pratique avec cette méthode, on se heurte souvent à une sous-détermination du problème ; la fonction de coût $z(\lambda)$ peut être insensible à certaines combinaisons de paramètres, nous avons une quasi singularité de l'opérateur Hessien tangent $[H] = [\partial^2 z/\partial\lambda_i\partial\lambda_j]$ qui donne des solutions reconstituant parfaitement les contraintes mesurées mais aberrantes en champs de pression.

Si on ne peut revoir la disposition des capteurs, on peut résoudre ce problème en réduisant le nombre des paramètres ; pour cela il est pratique d'utiliser une procédure automatique ou on se ramène à un vecteur de paramètre λ' de rang réduit lié à λ par la relation :

$$\lambda = [V]\,\lambda'$$

$[V]$ étant une matrice rectangulaire dont les colonnes sont les vecteurs propres du Hessien tangent initial $[H(\lambda = 0)]$ correspondant aux valeurs propres notablement différentes de zéro.

Dans les tests sur le Mirage 2000, on n'a pas pu dépasser un nombre de 5 paramètres réduit par effet rigide, et il est apparu que les résultats sont très sensibles aux pondérations relatives de chaque voie, ce qui a obligé à de nombreux tatonnements pour trouver une solution plausible.

En définitive, cette méthode de moindres carrés nous a semblé dépendre beaucoup trop de ces facteurs de pondérations subjectifs pour pouvoir être appliquée systématiquement.

3.7 - Deuxième technique : méthode d'optimisation sous contrainte

On pose le problème de l'identification sous la forme :

$$\text{minimiser} \quad z(\lambda) = \int \left\{ \left[\frac{\partial Kp}{\partial qr}\right] \text{théorique} - \left[\frac{\partial Kp}{\partial qr}(\lambda)\right] \right\}^2 ds$$

en satisfaisant aux inéquations

$$\frac{\partial\sigma k}{\partial qr}\text{ mes.} - \varepsilon_k < \frac{\partial\sigma k}{\partial qr}(\lambda) < \frac{\partial\sigma k}{\partial qr}\text{ mes.} + \varepsilon_k$$

ce qui signifie qu'on cherche une solution la plus proche possible de la théorie tout en restituant les contraintes mesurées dans une fourchette donnée.

La minimisation sous contrainte est menée par une méthode d'optimisation quadratique dite de "gradient sphérique" qui est présentée en annexe.

Cette formulation pallie aux principaux inconvénients de la méthode des moindres carrés.

- Il n'y a pas d'indétermination possible du problème, en l'absence d'observations, la procédure complète automatiquement par la théorie.

- Les paramètres de réglages sont <u>objectifs</u>, ce sont les incertitudes ε sur la reconstitution de chaque mesure ; on les évalue à <u>partir de</u> la précision des mesures et du prélissage et surtout de celle du calcul des contraintes, revélée par les correllations de l'étalonnage.

En pratique, il arrive que l'algorithme détecte qu'il n'y a pas de solution admissible compte tenu des contraintes qu'on s'est données ; cela se produit quand la grille de recalage est trop grossière vis-à-vis de l'exigence des contraintes à restituer ; la solution est de raffiner la grille de recalage.

L'obtention de champs aberrants traduit des erreurs au niveau des mesures ou de l'étalonnage ; on a une bonne indication des contraintes erronées par l'analyse des multiplicateurs de Lagrange de la solution (ils traduisent l'influence sur la fonction de coût de la fourchette qu'on s'est donnée sur les contraintes en butée).

3.8 - Test sur le Mirage 2000

Nous avons évalué les possibilités de la méthode sur le Mirage 2000-01 ; nous présentons ici quelques points caractéristiques de l'étude illustrant l'identification des effets d'incidence et de braquage de gouverne pour des stimulus symétriques :

- 3 stimulus à Mach 0,7 (4800 ft, 14000 ft et 25200 ft),
- 3 stimulus à Mach 1,5 (25600 ft, 40500 ft et 49900 ft).

Nous avons tracé planches 7 et 8 les reconstitutions et les mesures des jauges sur quelques voies significatives.

Nous comparons les champs de pression des effets d'incidence et de braquage des élevons "rigides" identifiés par les stimulus de badin différents aux mêmes Mach (Mach 0,7 planches 9 et 10, Mach 1,5 planches 11 et 12) ; nous avons utilisé l'option de recalage des effets souples.

Les champs de pression obtenus par des stimulus à différentes altitudes sont semblables et ils diffèrent quelque peu des champs théoriques ; les distorsions des champs reconstitués dans la zone d'emplanture ne sont probablement pas aérodynamiquement réalistes.

Pour traduire en charges "classiques" les résultats, nous présentons les reconstitutions comparées des courbes d'efforts tranchants par g de facteur de charges (Mach 0,7, z = 1000 ft, Mach 1,5, z = 25000 ft).

La relative insensibilité des résultats à l'altitude du stimulus de recalage, est due au fait qu'ils ont été tous effectués dans la plage d'incidence où l'aérodynamique reste assez bien linéaire.

4 - CONCLUSIONS ET DEVELOPPEMENTS

Notre procédure d'identification est efficace en utilisant les stimulus dans les zones où l'aérodynamique est linéaire ; elle nécessite une bonne qualité pour le modèle élastique éléments finis recalé par les étalonnages au sol, car on ne peut exiger plus de précision pour la reconstitution des mesures en vol qu'on est capable d'en obtenir au sol avec des chargements connus exactement ; il faut particulièrement faire attention avec les parties mobiles (becs, gouvernes), surtout quand elles sont liées hyperstatiquement au caisson.

Les champs de pression reconstitués sont sensibles au nombre et à la disposition des jauges ; on ne doit pas trop s'attacher à leur réalité du point de vue aérodynamique ; l'important pour le calcul des charges est qu'ils restituent valablement, au travers du modèle, les contraintes observées en vol.

C'est dans cet esprit que nous développons la méthode pour les zones où l'aérodynamique est non linéaire (grandes incidences, transsonique) ; l'identification porte alors simultanément sur les résultats de plusieurs manoeuvres, elle fournit les coefficients de recalage pour des successions d'états des paramètres aérodynamiques et non pour des effets de pente.

Dans son principe la méthode peut s'étendre à l'identification des effets d'aérodistorsion et des effets d'aérodynamique instationnaire, la principale difficulté est de disposer alors de manoeuvres où ces effets soient "observables", l'exigence de validation préalable du modèle élastique dynamique est évidemment renforcée.

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ANNEXE

A-1 - EQUATIONS DE BASE DU COUPLAGE AEROELASTIQUE

A-1.1 - Phases préalables (Batch)

- Aérodynamique théorique : champs de pression discrétisés

  $Kp = Kpo + [\frac{\partial Kp}{\partial qr}]qr + [\frac{\partial Kp}{\partial qs}]qs$

  q_r : effets aérodynamiques rigides (incidence, braquage de gouvernes, etc...)

  q_s : déformée des surfaces portantes exprimées dans une base de monômes

  (q_r et q_s de rang quelques dizaines)

- Equation du mouvement de l'avion quasi rigide

  $Fc.d.g. = \frac{1}{2}\rho V^2 (Co + [Cr]qr + [Cs]qs)$

  Les coefficients aérodynamiques [Cr], [Cs] et Co sont les torseurs résultants des champs de pression.

- Chargement du modèle Eléments finis

  $F = [P]f$ $[P] = [Paero, Pinertie, Poutres]$ chargements de base

- Résolution éléments finis (sur C.L. isostatique)

  $X = [K]^{-1} F \rightarrow$ déformées de base $[B] = [K]^{-1} [P]$

- Opérateurs "concentrés"

  . Contraintes et efforts aux points sensibles $[\partial\sigma/\partial f] = [\partial\sigma/\partial X]\,[B]$

  . Torseur résultant et mouvement rigide $[\partial Fcdg/\partial f], [\partial Rcdg/\partial f]$ (pour chaque masse de base)

  . Chargement aérodynamique décomposé dans les chargements de base

  $faero = [R]\,Kp \rightarrow [\partial f/\partial qr] = [R]\,[\partial Kp/\partial qr], [\partial f/\partial qs] = [R]\,[\partial Kp/\partial qs]$

  . Lissage des déformées E.F. dans les formes aérodynamiques

  $qs = [L]\,X \rightarrow [\partial qs/\partial f] = [L]\,[B]$

A-1.2 - Phases répétées (intéractives)

- Recalage empirique (identification sur la soufflerie ou le vol)

  . Effets "rigides"

  $[\partial Kp/\partial qr]recal\acute{e} = [\partial Kp/\partial qr]th.[I + [\phi]\lambda]$

  λ : paramètres de recalage

  $[\phi]$: fonction d'interpolation de la grille de recalage (voir § 3.5)

  $\rightarrow [\partial f/\partial qr]recal\acute{e} = [R]\,[\partial Kp/\partial qr]recal\acute{e}$

  . Effets "souples"

  $[\partial f/\partial qs]recal\acute{e} = [\partial f/\partial qs]th. + ([\partial f/\partial qr]recal\acute{e} - [\partial f/\partial qr]th.)\,[\alpha]$

- Calcul des charges d'inertie dans les configurations envisagées (combinaisons linéaires des effets d'inerties élémentaires)

  $\rightarrow [\partial f/\partial \gamma]$

- Charge aérodynamique de base équilibrée par les effets d'inerties

  $\rightarrow [\partial f/\partial qr]\acute{e}quil. = [I - \partial f/\partial\gamma]\,[M]^{-1}[\partial Fcdg/\partial f]\,[\partial f/\partial qr]recal\acute{e}$

  $[\partial f/\partial qs]\acute{e}quil. = [I - \partial f/\partial\gamma]\,[M]^{-1}[\partial Fcdg/\partial f]\,[\partial f/\partial qs]recal\acute{e}$

- Calcul des coefficients aérodynamiques avion souple

Elimination des effets souples

$qs = \frac{1}{2} \rho V^2 \, (Ao + [A1]qr + [A2]qs) + A_{autre}$

$[A1] = [\partial qs/\partial f] \, [\partial f/\partial qr]_{équil.} \quad [A2] = [\partial qs/\partial f] \, [\partial f/\partial qs]_{équil.}$

(plus correction du mouvement rigide)

$\rightarrow qs = \frac{1}{2} \rho V^2 \, ([\mu]_{qr} + \mu_o) + \mu_{autre}$

$[\mu] = [D]^{-1}[A1] \quad [D] = (I - \frac{1}{2} \rho V^2 \, [A1])$

La singularité de $[D]$ exprime la divergence statique

. Coefficient aérodynamique apparent

$[C] \quad [Cr] + \frac{1}{2} \rho V^2 \, [Cs] \, [\mu]$

$Co = Co_r + [Cs] \, (\frac{1}{2} \rho V^2 \, \mu o + \mu_{autre})$

- Contraintes et efforts unitaires aéroélasticité éliminée

$[\partial\sigma/\partial qr] = [\partial\sigma/\partial f] \, ([\partial f/\partial qr] + \frac{1}{2} \rho V^2 \, [\partial f/\partial qs] \, [\mu])$

- Calcul de manoeuvre avion quasi rigide

. Résolutions des équations de la mécanique du vol avec $[C] \rightarrow qr(t)$

. Contrainte et effort en manoeuvre aux points sensibles

$\sigma(t) = \sigma o + \frac{1}{2} \rho V^2 \, [\partial\sigma/\partial qr] \, qr(t)$

A-2 - DERIVATION PAR RAPPORT AUX PARAMETRES DE RECALAGE

Pour aider la compréhension, nous précisons les indices de chaque opérateur de base, soit :

$fm, Kp_n, qr_j, qs_{j'}, \phi_{n,i}, \lambda_{i,j}$

- Dérivées des effets rigides

$[\frac{\partial^2 fm}{\partial qrj \; \partial\lambda i,j}] = [Rm,n] \, [\frac{\partial Kpn}{\partial qrj}]th.[\phi n,i]$

torseur résultant $\rightarrow$ dérivées coef. Aéro. rigide $[\frac{\partial Cr}{\partial\lambda ij}]$

- Dérivées des effets souples $[\frac{\partial^2 fm}{\partial qsj' \; \partial\lambda i,j}] = [\frac{\partial^2 fm}{\partial qrj \; \partial\lambda i,j}] \, [\alpha j,j']$

torseur résultant $\rightarrow$ dérivées coef. Aéro. souple $[\frac{\partial Cs}{\partial\lambda i,j}]$

- Dérivées des effets rigides et souples équilibrées par les charges d'inertie induites : même procédure que pour les effets eux-mêmes

- Différentiation de l'élimination des effets souples

$$dqs = \frac{1}{2}\rho V^2 ([dA1]qr + [dA2]qs + [A2]dqs)$$

$$dqs = [D]^{-1} ([dA1] + [dA2] [\mu])qr$$

$$\rightarrow [d\mu] = [D]^{-1} ([dA1] + [dA2] [\mu])$$

soit :

$$[\frac{\partial \mu}{\partial \lambda ij}] = [D]^{-1} [\frac{\partial qs}{\partial f}] ([\frac{\partial^2 fm}{\partial qrj\, \partial \lambda ij}] + [\frac{\partial^2 fm}{\partial qsj\, \partial \lambda ij}] [\mu])$$

- Dérivées des coefficients aérodynamiques apparents

$$[\frac{\partial C}{\partial \lambda i,j}] = [\frac{\partial Cr}{\partial \lambda i,j}] + \frac{1}{2}\rho V^2 ([\frac{\partial Co}{\partial \lambda i,j}] [\mu] + [Co] [\frac{\partial \mu}{\partial \lambda i,j}])$$

- Dérivées des contraintes unitaires aéroélasticité éliminée

$$[\frac{\partial^2 \sigma}{\partial qr \partial \lambda i,j}] = [\frac{\partial \sigma}{\partial f}] ([\frac{\partial^2 fm}{\partial qr \partial \lambda ij}] + \frac{1}{2}\rho V^2 ([\frac{\partial^2 fm}{\partial qs \partial \lambda ij}] [\mu] + [\frac{\partial f}{\partial qs}] [\frac{\partial \mu}{\partial \lambda ij}])$$

A-3 - ALGORITHME D'IDENTIFICATION

- 1ère méthode : moindre carré

minimisation de

$$z(\lambda) = \sum_{\text{voies de mesures}} \pi_K \left(\frac{\partial \sigma K}{\partial qr} \text{mes} - \frac{\partial \sigma K}{\partial qr}(\lambda)\right)^2$$

méthode de Pseudo Newton

- départ $\lambda = 0$
- gradient $G = [\partial z/\partial \lambda] = \sum_{\text{voies de mesures}} \pi_K [\frac{\partial \sigma mes}{\partial qr} - \frac{\partial \sigma(\lambda)}{\partial qr}] [\frac{\partial^2 \sigma}{\partial qr\, \partial \lambda ij}]$
- pseudo Hessien $[H] = \sum_{\text{voies de mesures}} \pi_K [\frac{\partial^2 \sigma}{\partial qr\, \partial \lambda i,j}] [\frac{\partial^2 \sigma}{\partial qr\, \partial \lambda i,j}]$

$$\lambda = \lambda + [H]^{-1} G$$

- 2ème méthode : optimisation sous contrainte

$$\Sigma \lambda ij^2 \quad \text{minimum}$$

+ inéquation pour chaque voie de mesure

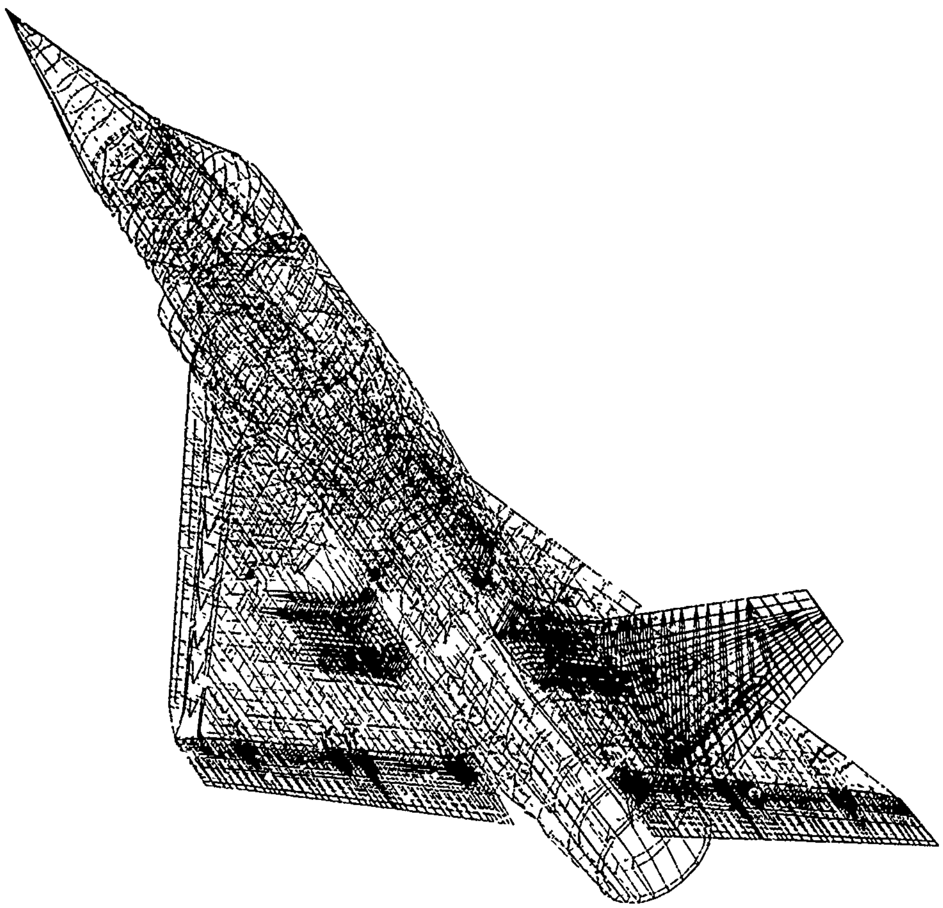
$$\frac{\partial \sigma K}{\partial qr} \text{mes} - \varepsilon_K < \frac{\partial \sigma K}{\partial qr}(\lambda i,j) < \frac{\partial \sigma K}{\partial qr} + \varepsilon_K$$

résolu par itération d'optimisation quadratique, $\frac{\partial \sigma K}{\partial qr}(\lambda i,j)$ étant linéarisé à chaque itération (convergence en 2 à 3 itérations).

Pl. 1

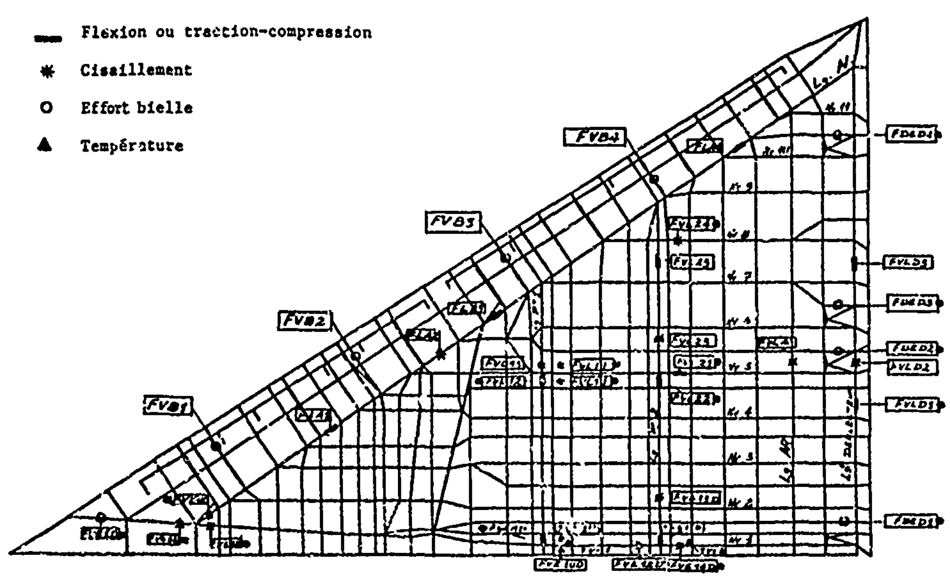
MIRAGE 2000

MODELE ELEMENTS FINIS



Pl. 2

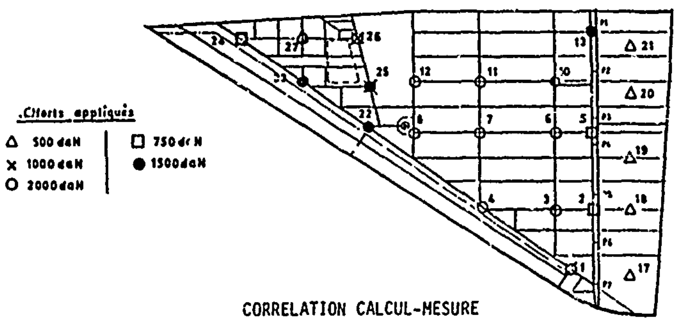
VOILURE MIRAGE 2000-01



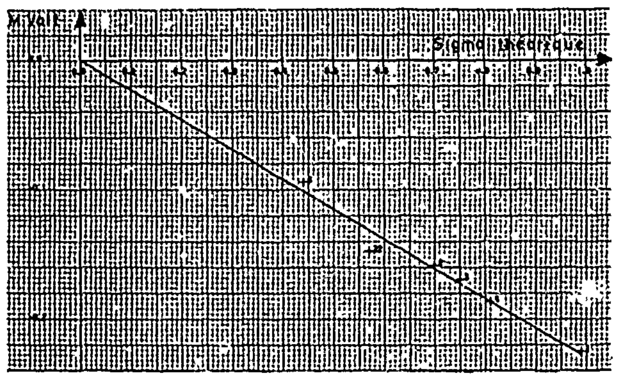
SCHEMA D'IMPLANTATION DES JAUGES

Pl. 3

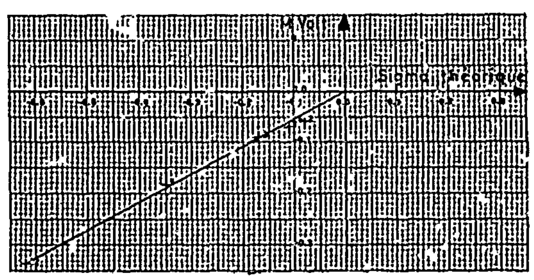
POSITION DES CHARGEMENTS D'ETALONNAGE



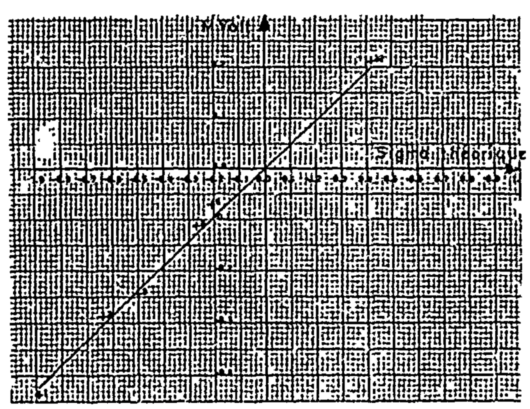
CORRELATION CALCUL-MESURE



FVE8D



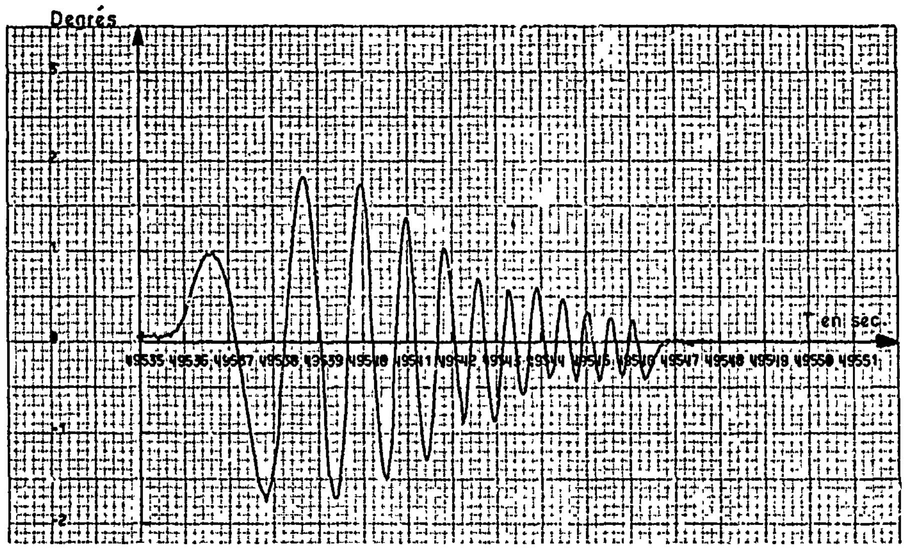
FVL13



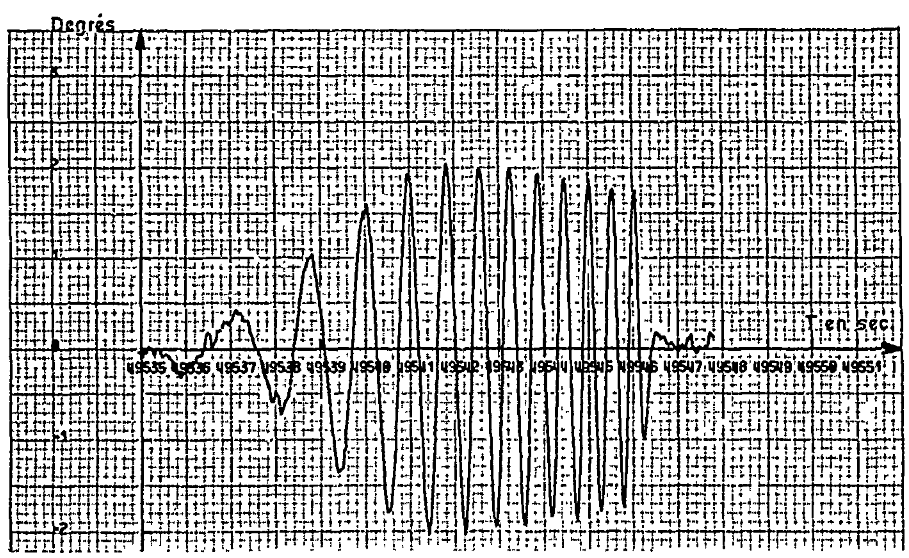
FVL14

MANOEUVRE DE STIMULUS

(Mach = 0,7   z = 14000 ft)



EVOLUTION DE L'INCIDENCE

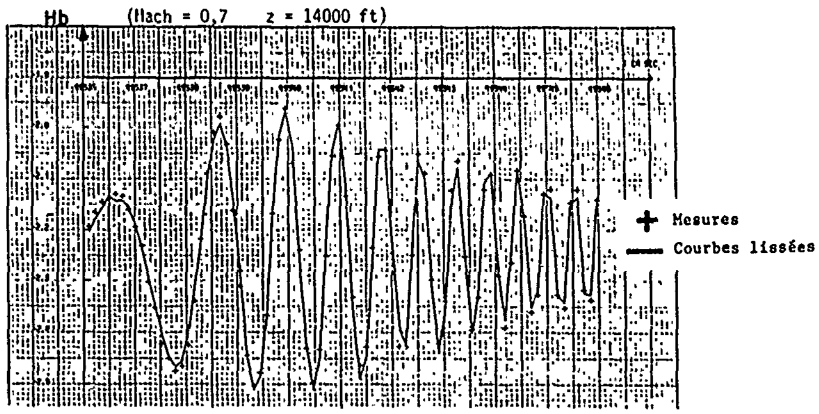


EVOLUTION DU BRAQUAGE DES ELEVONS

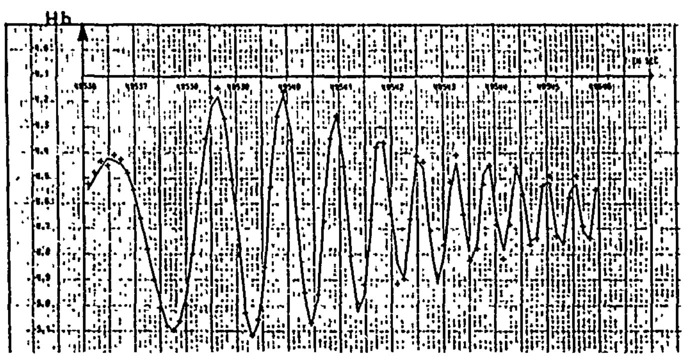
Pl. 5

LISSAGE DES CONTRAINTES MESUREES DANS UN STIMULUS
(Mach = 0,7   z = 14000 ft)

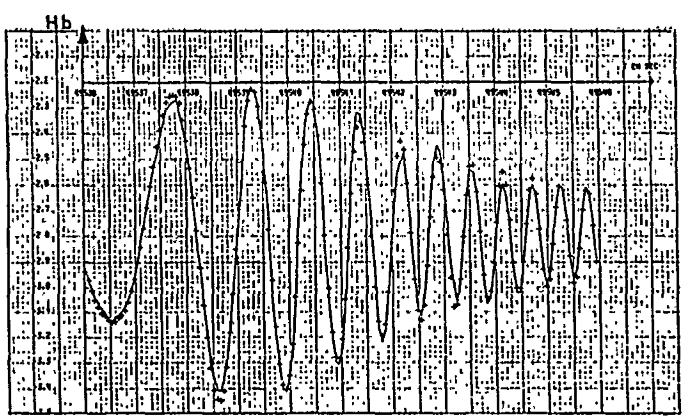
FVLD 1



FVLD3



FVL 13



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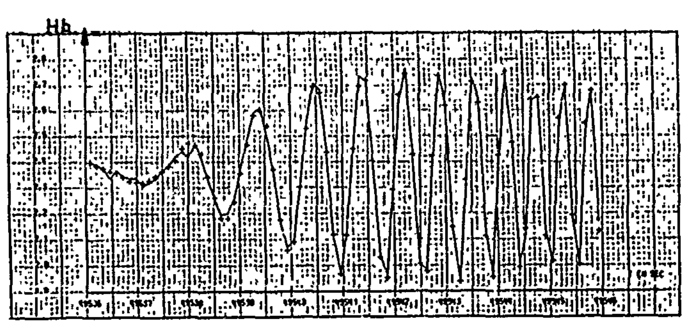
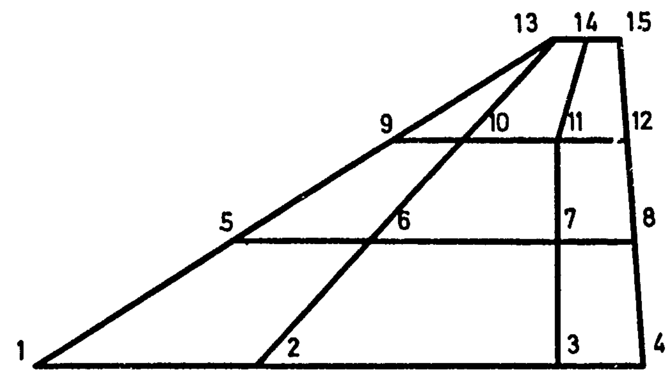
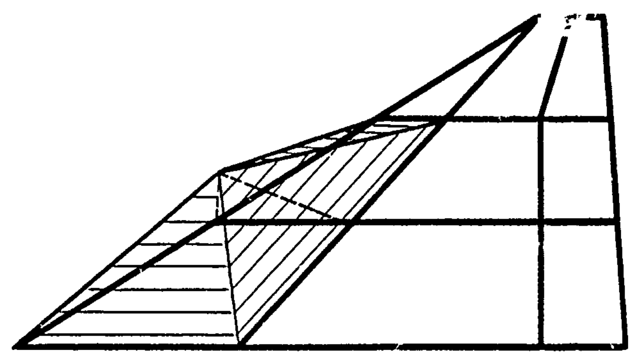


Pl. 6

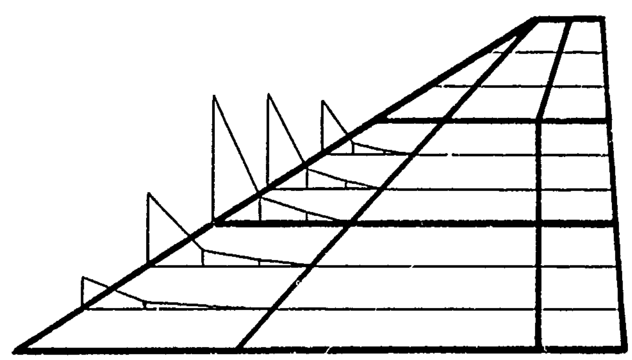
DEFINITION DES PARAMETRES DE RECALAGE



GRILLE DE RECALAGE



CHAMPS DE CORRECTION DU PARAMETRE $\lambda 5$

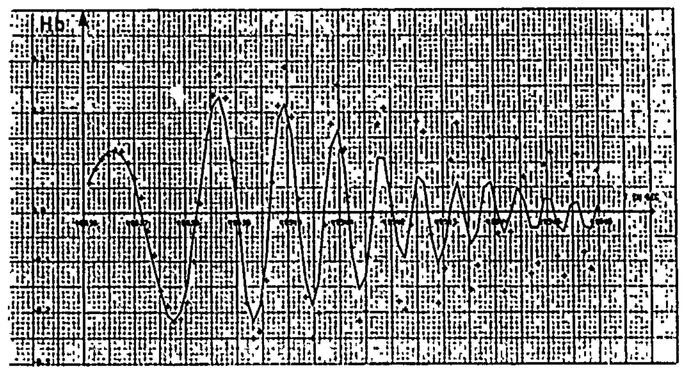


INFLUENCE DU PARAMETRE $\lambda 5$
SUR UN CHAMP D'INDIDENCE

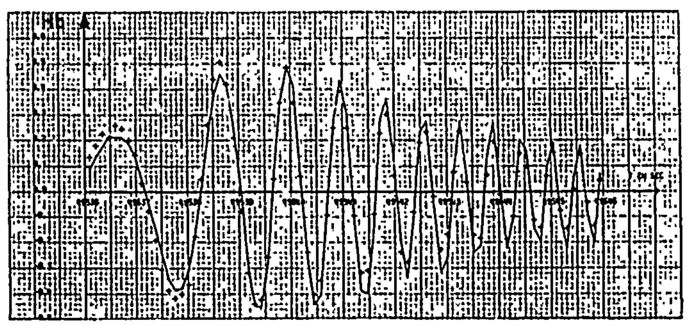
REPONSE DFS JAUGES
(Mach = 0,7   z = 14000 ft)
COMPARAISON THEORIE-MESURES

+ Mesure
— Théorie

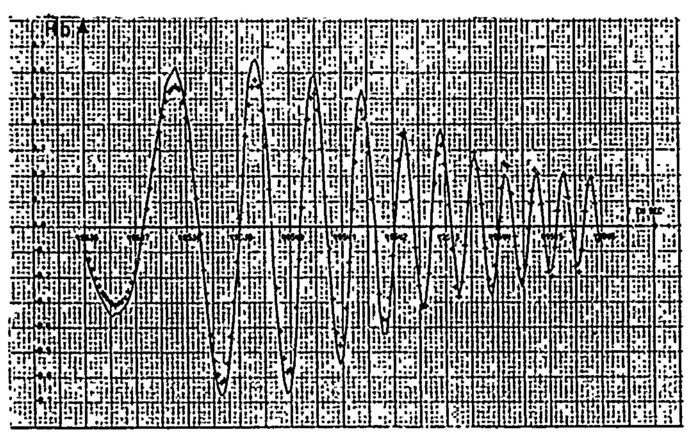
FVL D1



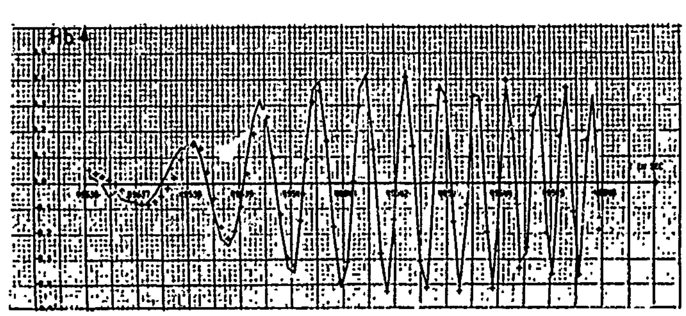
FVL D3



FVL 13



FVL 14



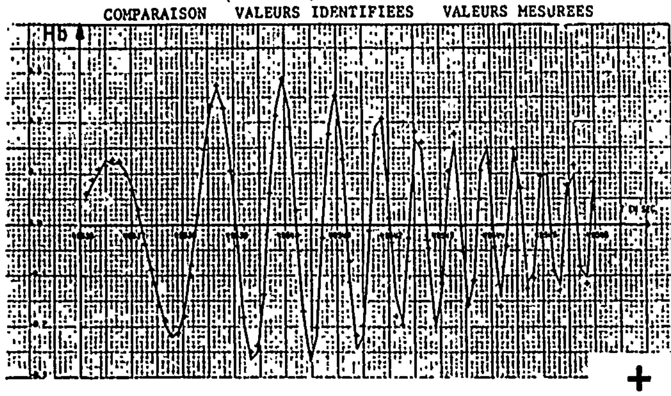
Pl. 8

REPONSE DES JAUGES

(Mach = 1,5   z = 14000 ft)

COMPARAISON VALEURS IDENTIFIEES VALEURS MESUREES

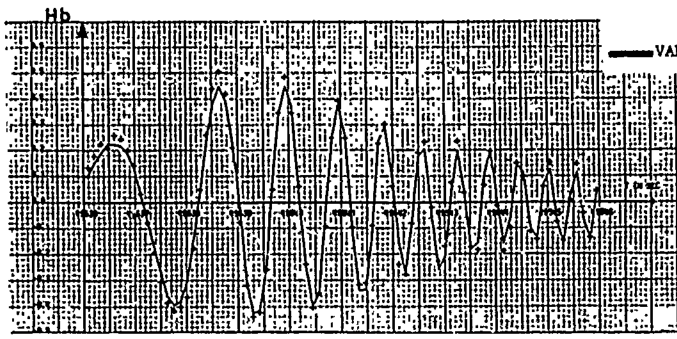
FVL D1



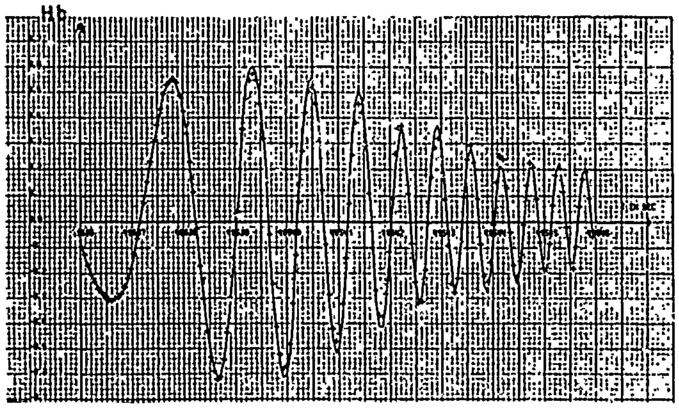
+ VALEURS MESUREES

— VALEURS IDENTIFIEES

FVL D3



FVL 13

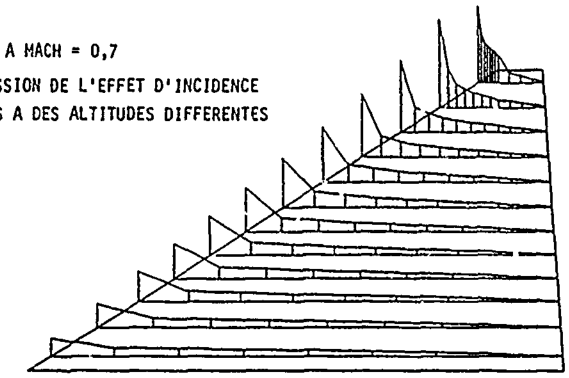


FVL 14

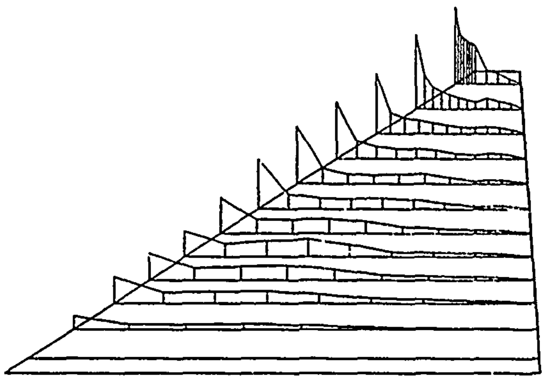
Pl. 9

IDENTIFICATION A MACH = 0,7

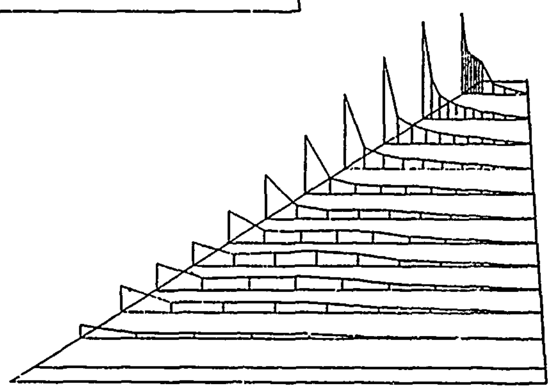
COMPARAISONS DES CHAMPS DE PRESSION DE L'EFFET D'INCIDENCE "RIGIDE" PAR 3 STIMULUS EFFECTUES A DES ALTITUDES DIFFERENTES



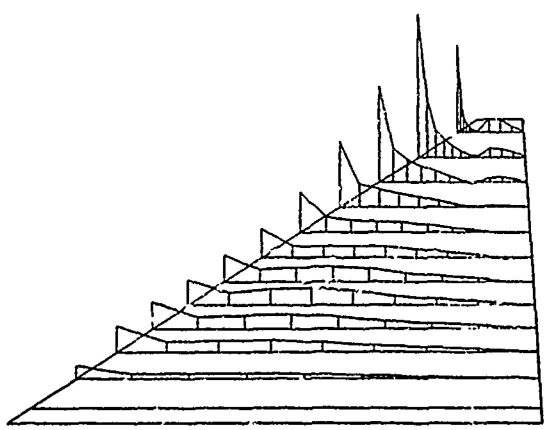
THEORIQUE



4800 ft



14000 ft



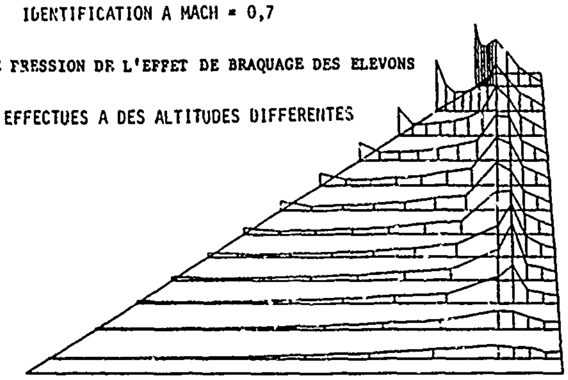
25200 ft

Pl. 10

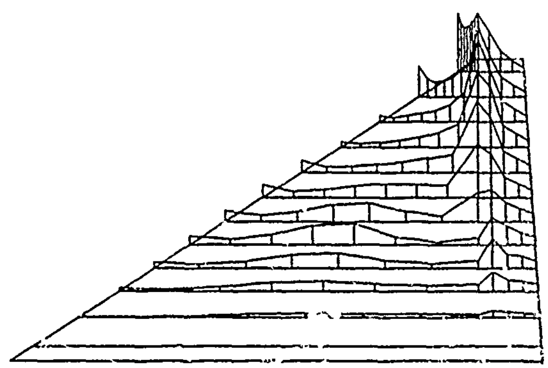
IDENTIFICATION A MACH = 0,7

COMPARAISONS DES CHAMPS DE PRESSION DE L'EFFET DE BRAQUAGE DES ELEVONS

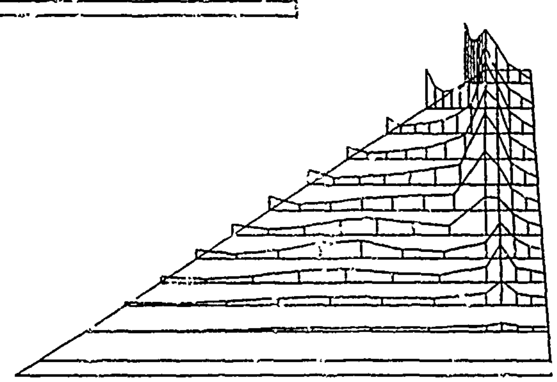
"RIGIDE" PAR 3 STIMULUS EFFECTUES A DES ALTITUDES DIFFERENTES



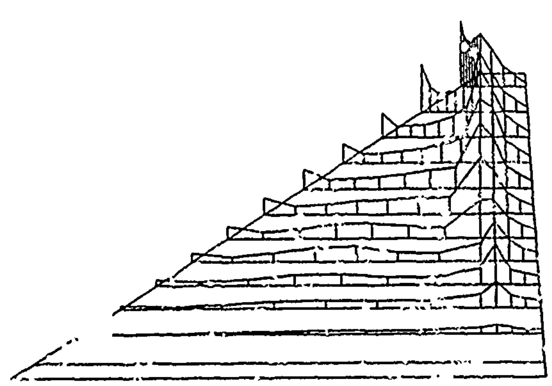
THEORIQUE



4800 ft



14000 ft

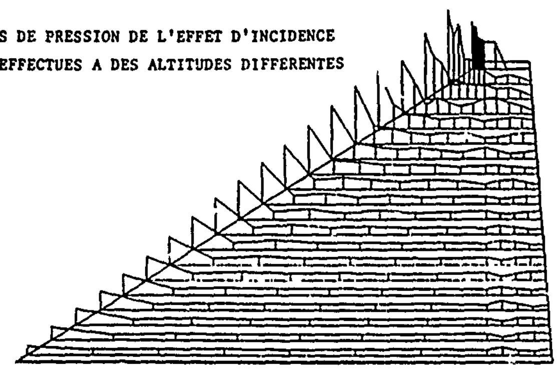


25200 ft

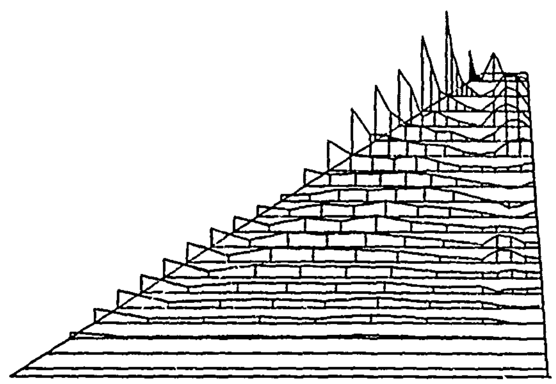
Pl. 11

IDENTIFICATION A MACH = 1,5

COMPARAISONS DES CHAMPS DE PRESSION DE L'EFFET D'INCIDENCE
"RIGIDE" PAR 3 STIMULUS EFFECTUES A DES ALTITUDES DIFFERENTES



THEORIQUE



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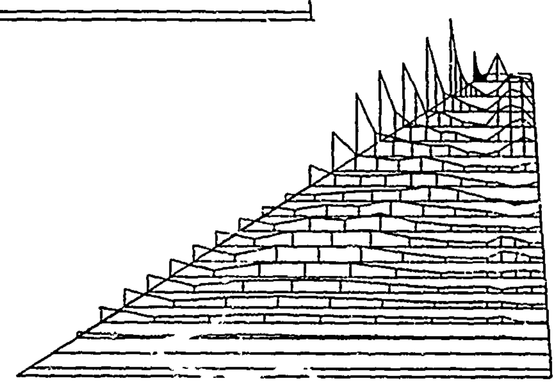

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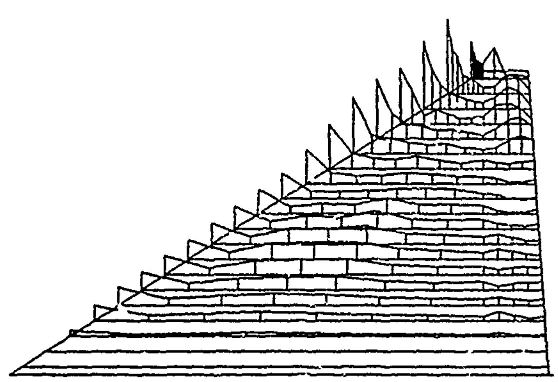


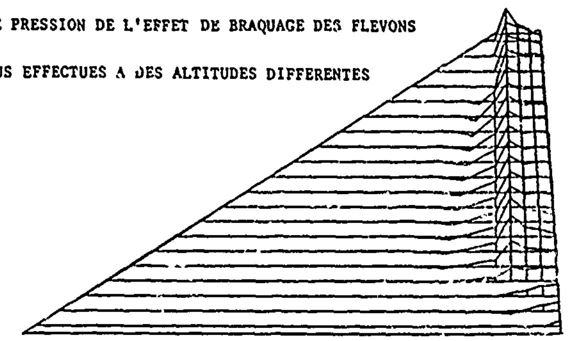
49900 ft

Pl. 12

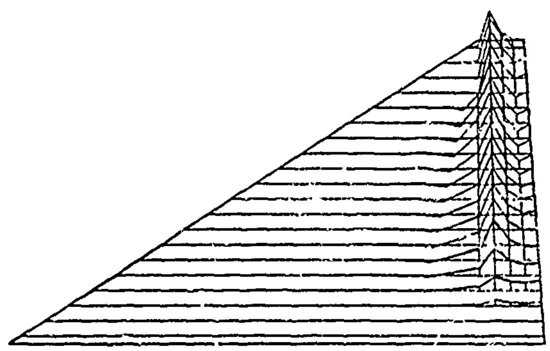
IDENTIFICATION A MACH = 1,5

COMPARAISONS DES CHAMPS DE PRESSION DE L'EFFET DE BRAQUAGE DES FLEVONS

"RIGIDES" PAR 3 STIMULUS EFFECTUES A DES ALTITUDES DIFFERENTES



THEORIQUE



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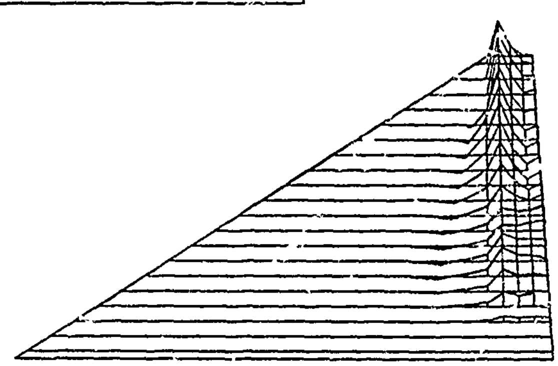

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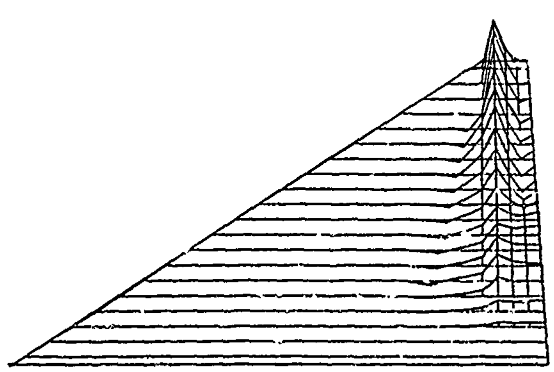
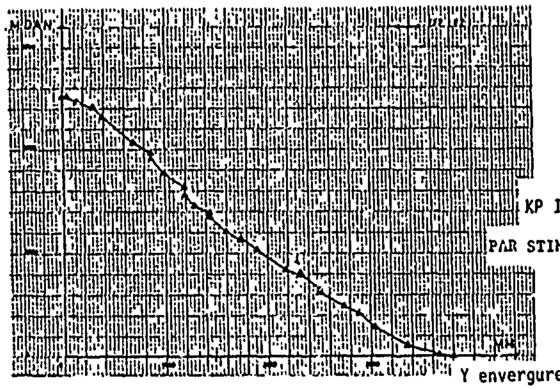
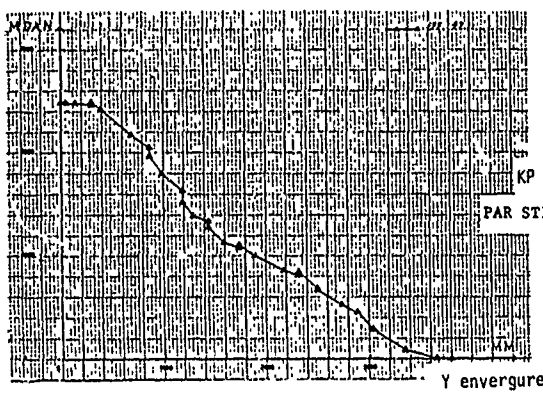


49900 ft

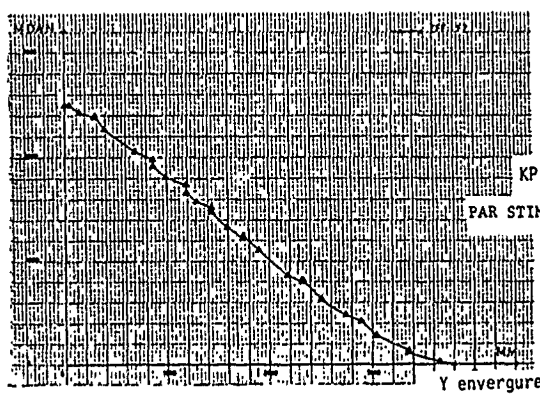
EFFORT TRANCHANT/G DE FACTEUR DE CHARGE (Mach 0,7   1000 ft)



KP IDENTIFIES
PAR STIMULUS   A - 4835 FT



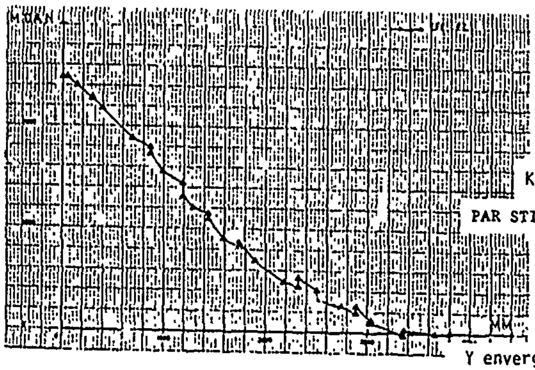
KP IDENTIFIES
PAR STIMULUS   A - 13974 FT



KP IDENTIFIES
PAR STIMULUS   A - 25183 FT

Pl. 14

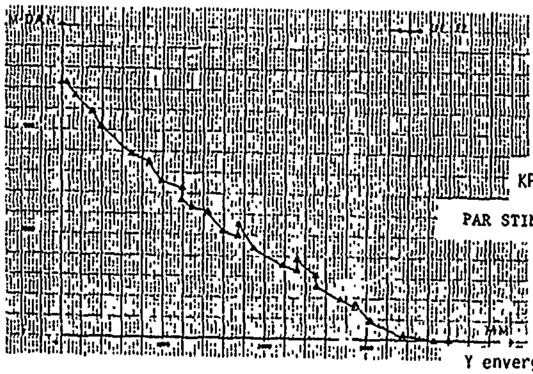
EFFORT TRANCHANT/G DE FACTEUR DE CHARGE (Mach 1,5 25000 ft)



KP IDENTIFIES

PAR STIMULUS A - 25593 FT

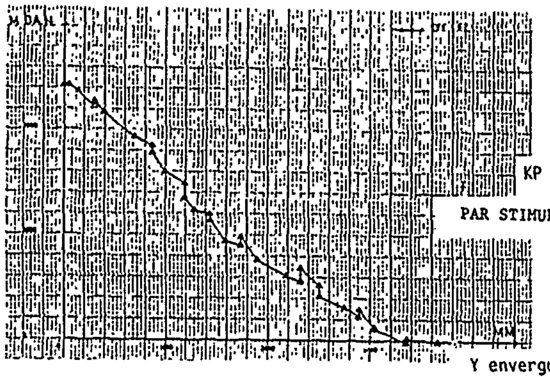
Y envergure



KP IDENTIFIES

PAR STIMULUS A - 40508 FT

Y envergure



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SESSION III - USE OF OPERATIONAL LOADS DATA FOR DESIGN AND TEST PURPOSES

SUMMARY RECORD

by
L Baranes
Service Technique des Programmes Aeronautiques,
Paris, France

In discussion and questions on paper No 13, it was observed that the method used was intended to derive extreme values of various parameters, but the analysis was based on a limited number of flights. This limitation raised the question of the basis on which extreme values were extrapolated from the normalised parameter profiles, and also the question of the probability levels assigned to extreme (ie design limit) load cases. One speaker pointed out an anomaly in the analysis in one case, where the start of pitch rate build-up appeared to precede application of elevator control. The design loads predicted were lower in some instances than MIL SPEC - 8861 requirements; if the philosophy proposed was accepted and developed for design use, the outcome would be some amendment to or variation from the MIL SPEC cases.

On paper No 14, questions were posed on the large (70%) exceedence observed in the rear fuselage lateral shear load compared with the design limit load estimate, and particularly on whether the latter was based on wind tunnel tests. The design load case figure was in fact based on calculation, supported by initial flight test results. No fatigue or damage tolerance data was available at the time flight testing was carried out. This was not the only case where the flight test programme showed loads in excess of theoretical limit load. However, static tests at Canadair showed that the structural design was generally conservative and it was decided not to carry out further verification work or amend limit load data.

In answer to questions on paper No 15, the author stated that loads were derived for the rigid aircraft case, although for full precision aeroelasticity should be considered for transient cases. Development work was in hand to take flexibility into account for rapid manoeuvres.

In general discussion, it was pointed out that there was a considerable difference between civil transport flying, where limit load conditions were extremely rare, and military operations, where the extremes of the intended flight envelope can be approached, and possibly even exceeded, quite regularly. On current aircraft the existence of appreciable Reserve Factors often prevented proof stress being reached or exceeded; but this situation might change for the next generation of aircraft. Occasional exceedences of 9g (normal load factor) on F-18 were quoted; although the aircraft was designed for 7.5g limit in standard configuration, 6-9 such exceedences in 5000 flying hours seemed acceptable, although it is difficult to check fully what damage such excursions may have caused. In the case of the F-16, the aircraft is designed for 10.5g and manoeuvres up to 9g occur frequently; however, the flight control system imposes a manoeuvre limitation on pilot demand.

Differences between UK practice (never exceed limits, coupled with a lower clearance for normal operations) and US practice (never exceed limits only) were mentioned. There appeared to be a clear need to identify any divergence between the authorised clearances and actual practice in service units, particularly for highly manoeuvrable aircraft which frequently carry external stores.

Aircraft using active control technology (ACT) may well experience very high manoeuvre rates, which could have significant fatigue as well as static load implications. It was necessary to establish both general and local structural effects of ACT, including those of unconventional manoeuvres, and ensure that the ACT system did generally function in service as designed. It was admitted that communication between flight control and loads specialists was not always fully effective. Considering the approach described in Paper No 13, the question of identifying standardised manoeuvres for future ACT aircraft needs to be addressed further. The Mirage 2000 system had an 8g system limit which could only be overcome by the pilot using considerable effort; manoeuvres approaching 8g were frequent, an effect recognised and taken account of in describing the design load spectrum. The loads imposed by the system itself, rather than by pilot input, had been allowed for in design, including design for fatigue cases. Reference was made to taileron loads on F-15 and Tornado, where it was believed that manoeuvring or terrain following might generate either high pilot input loads or lower-level stabilisation system-induced loads more often than predicted in design spectra.

TORNADO - STRUCTURAL USAGE MONITORING SYSTEM (SUMS)

A P WARD
Deputy Chief Stressman
British Aerospace PLC
Warton Division
Preston PR4 1AX
England

SUMMARY

The paper reviews the background for the establishment of the Tornado Structural Usage Monitoring System (SUMS). Following a brief discussion of the possible monitoring methods the decision to opt for two approaches using flight parameters and load calibrated strain gauges is amplified. The calibration and instrumentation requirements are described and the use of SUMS data is discussed.

1. INTRODUCTION

The Tornado is a multi role combat aircraft having variable geometry wings and moving tailerons in addition to the more conventional controls (Fig. 1). The derivation of fatigue load spectra was made using best estimates of likely usage and fatigue monitoring in service was centred around the use of the fatigue meter and flight-by-flight record sheets. Following a study of improved means of measuring the fatigue usage of the aircraft it was decided that the RAF would install a comprehensive fatigue monitoring system on a limited number of aircraft. The system is known as SUMS (Structural Usage Monitoring System).

The paper describes the background to the choice of the SUMS approach, summarises the instrumentation and calibration requirements and discusses the use of SUMS data.

2. BACKGROUND

The limitations of the fatigue meter (1) as a monitoring device are well enough known and the problems associated with estimation of fatigue load spectra have previously been described (2). During the late 1970s the Warton Division of British Aerospace undertook a study, on behalf of the Ministry of Defence, entitled "Investigation to determine the feasibility of establishing a new operational fatigue life monitoring system". The BAe proposal associated with this contract included three main objectives, these being that the system should:

(i) enable "fatigue life consumed" to be calculated for different components on individual aircraft thus making for efficient fleet management.

(ii) provide information of value in defining load spectra for new aircraft.

(iii) be capable of identifying any unusual manoeuvres and establish data for use in defining new static design requirements.

During the same period work on enhanced fatigue monitoring method for RAF Tornados also took place. As BAe Warton was one of the three partner companies of PANAVIA (the others being MBB, Munich and Aeritalia Turin), the designers and builders of the Tornado, it necessarily followed that both tasks were closely linked.

3. PHILOSOPHY

Three methods of approach were considered for monitoring fatigue usage. These consisted of:

(i) use of direct monitoring devices which measured local strain histories.

(ii) direct measurement of fatigue loading applied to the structure.

(iii) measurement of aircraft response parameters from which loads on various surfaces could be calculated.

The first method was quickly rejected on the grounds that a large number of devices would necessary to cover the full structure, and that these devices would not provide the type of information necessary to satisfy the second and third requirements referred to above (load spectra for new aircraft and data on unusual manoeuvres). Such devices would, however, probably provide a method for use in fleetwide monitoring on an aircraft-by-aircraft basis. A survey of these methods is given in reference 3.

The second method, which involved comprehensive strain gauging and calibration of each airframe structure, was, initially, not thought to be the most appropriate approach as

(i) suitable capacity recorders for use in a fully equipped service aircraft were not available.

(ii) there was concern about the reliability of strain gauge installations on a long term basis, in particular because of the possibility of damage during routine servicing.

(iii) each aircraft would require a full load calibration (4).

The third approach appeared to be the most attractive. This was based on the premise that, by using a limited number of accelerometers mounted on the aircraft at different locations to measure aircraft response and taking into account a small number of other parameters it would be possible to predict loadings at any location on the aircraft with a reasonable degree of accuracy. It was thought that the necessary recording capacity would be less than for the load calibrated strain gauge approach and that the instrumentation would be more reliable. There was the additional advantage that load calibration of each aircraft would not be necessary.

Practically all the effort spent by BAe on the research contract referred to earlier was concerned with an investigation of the parametric approach. This was a theoretical study that used a mathematical model of the Tornado (5) to generate load-time histories. These were then assessed, using multiple regression analysis techniques to establish load prediction equations using the minimum number of aircraft response parameters that would give an acceptable correlation with the fully deterministic model. This work is reported in references 3 and 6. Other work by RAE using measured aircraft response data on a Lightning aircraft is reported in references 7 and 8 with further comment on the method being given in reference 9.

The BAe theoretical study initially gave very good results based on an assessment of fin loads. However, the development of the model to cover other areas of the structure and to make allowance for differing stores carriage and use of manoeuvre devices with varying wing sweep led to increasingly complicated predictor equations. In addition considerable debate took place with regard to the representativeness of the theoretical manoeuvres used with the mathematical model in order to develop the predictor equations. Although the model had been proven to give reliable results based on flight test flying there was a body of opinion that believed that service pilots would find ways of using the aircraft which were quite different from any covered by the model. It was therefore felt that predictor equations should be developed from service flying rather than from the model.

After much discussion it was agreed that for the Tornado SUMS programme both the parametric approach and the load calibrated strain gauge approach would be followed, the latter taking preference, in terms of effort, prior to the calibrated aircraft entering service.

Seven aircraft were allocated to the SUMS programme, four of these being Strike aircraft and three being Trainers. An assessment of the most important structural assemblies led to the selection of 18 loading actions to be measured (Fig. 2), these being:

- wing bending moment at four stations
- wing shear at three stations
- wing torsion at three stations
- fin and taileron root shear force, bending moment and torsion
- front and rear fuselage vertical bending moment

In a number of cases measurements were required on both port and starboard sides of the aircraft thus increasing the number of channels required. From a study of these needs, and taking into account further instrumentation requirements to measure aircraft response and to enable damaging areas in each sortie to be identified, it was obvious that all the data could not be recorded on each aircraft. It was decided, that, for maximum flexibility in the programme, the full instrumentation fit would be installed and a full calibration performed on each of the seven aircraft. Each aircraft would then be assigned to a particular facet of the data recording exercise and the appropriate load measurement channels selected to match these requirements. It was provisionally decided that four aircraft would be assigned to the study of empennage loading and three to the study of wing and centre fuselage loads. For comparison purposes certain loads would be recorded on all aircraft.

4. INSTRUMENTATION REQUIREMENTS

During the early development of the Tornado a number of aircraft were used for test flight purposes, two of these being strain gauged and calibrated to measure loads. Experience gained from this exercise was taken into account in the SUMS programme, however significant compromises were necessary when applying load calibration approaches to service aircraft. The development aircraft required 75 channels for loading parameters, Table 1, and a further 63 channels for the computation of inertia loads and flight conditions, these covering rotational and translational velocities and accelerations, control surface positions, altitude, mach number, engine rotor speeds, temperatures etc. A recording capacity of 4096 samples per second was used whereas, for the service monitoring exercise, a Plessey recorder limited to 512 samples per second, was used.

As noted in section 3, the actual data to be recorded on each aircraft differs but, as an example, Table 2 illustrates the requirements for all aircraft in service. The first aircraft will have, as its primary aim, the objective of determining empennage loading, the strain gauge channels to be monitored being those shown on Table 3.

The aircraft instrumentation requirement is illustrated, in block diagram form, on Fig. 3. Note that although nominally 40 signals of strain gauge data are available only 16 will be recorded on any single aircraft.

In order to arrive at the loading measurements required a total of 64 strain gauge bridges were used. As a result of the work on the first aircraft this number has been reduced to 57 bridges for the third and subsequent aircraft. In addition there are three "hot spot" gauges installed for direct comparison with similar strain gauges in the major airframe fatigue test.

5. CALIBRATION

The load calibration of each aircraft was accomplished using a purpose built loading frame into which each aircraft was positioned (Fig. 4). Loads were applied to existing aircraft hard points (such as engine mountings) or by means of contour boards on the lifting surfaces.

Prior to applying each loading case the structure was exercised by applying the calibration loads a minimum of three times. Strain gauge responses at the zero and maximum load levels were recorded in each case. The actual calibration loads were then applied in increments of 10% up to the maximum load before unloading in 20% decrements. Each calibration load case was repeated three times. Table 4 provides a summary of the calibration loading.

Strain gauge responses and load cell output at each load level were processed on a VAX-11 computer in order to identify any instrumentation or loading problems at an early stage.

Wing and fin strain gauge bridges were electrically combined after the first 16 and 21 wing and fin load cases respectively in order to reduce the number of parameters to be recorded. The signals from the combined bridges were subsequently recorded on the ground logging system for a further 21 wing cases and 21 fin cases. Finally, at the end of the calibration, four cases were repeated with the selected bridge responses recorded on the on-board SUMS system.

Equations to relate loading at each of the monitoring stations to strain gauge responses were then derived using regression techniques.

6. DATA INTEGRITY

Based on the improved quality of data obtained in the Jet Provost empennage studies compared with the Jaguar data (ref. 10) and on the good data obtained from Nimrod studies performed by BAe Woodford, coupled with continuing improvements in instrumentation it was decided that the only data integrity checks to be performed would be the synchronisation count, checks for parity errors and single point drop out. However this decision was to be reviewed following the examination of data from the shakedown flights.

Instrumentation checks were planned to take place throughout the programme. These include checks of amplifier sensitivities and recalibration of transducers along with frequent (weekly) checks of datum values.

The assessment of the importance of datum drift will take place retrospectively. There is no correction for datum drift on the software, however, pre and post flight values will be determined and output for further analysis. Initially these data will be compared with fuel states and stores configurations.

EMC checks were included as part of the checks on the aircraft system.

7. DATA COMPRESSION

The large quantity of data generated by the Tornado SUMS programme requires steps to be taken to reduce the analysis time as much as possible. The Jet Provost studies referred to earlier (10) highlighted problems in analysis cost and time. To reduce these costs hard copy traces of each time history were examined to eliminate "inactive" time slices prior to subsequent computer analysis of the data. In order to reduce the amount of manual intervention required in the analysis process in-flight data compression is built-in to the Tornado system. One method of achieving such a reduction in data was that described by van Dijk and Nederveen in their paper on the DEMON system (ref. 11) whereby a range-filtering process was used such that "the load events detected are all peaks and valleys associated with a change of at least a specified range - threshold value". Only these peaks and valleys were recorded (in their chronological sequence) thus giving a significant reduction in the amount of data stored.

A different approach is being followed on Tornado which relies on identifying periods of inactivity in-flight. The system, which has been developed by RAE Farnborough, relies on the use of a buffer store prior to recording the flight data. The system switches between high and low sampling rates dependent upon the degree of activity sensed by a number of transducers situated on the aircraft. During low level activity the sampling rate is only 1 sps. If any of the transducers are triggered by increasing loading activity then the system can take data from two seconds before the event to four seconds after the event at the planned fast sampling rates (Table 2).

8. DETERMINATION OF LOCAL STRESSES FROM OVERALL LOADS

The merit in measuring fatigue structural loading rather than local stress histories is that it should be possible to establish stress histories at any location on the airframe from the overall loading. In specifying a load monitoring programme on an aircraft it is not necessary to know the fatigue critical locations beforehand whereas a stress monitoring survey relies on the need to know the areas of concern.

Studies have been performed to demonstrate the way in which stress histories can be derived from a knowledge of the overall loading histories. For a wide range of load cases overall loading was determined and local element stresses established from a NASTRAN finite element analysis of the structure. For particular elements algorithms to predict local stresses from overall loads were determined by regression analysis. Correlation coefficients of between 0.97 and 1 were achieved for a representative number of cases covering taileron, fin and wing loading actions (Table 5).

9. SHAKEDOWN FLIGHTS

Prior to return to service a shakedown flight is necessary on each of the seven aircraft in order to provide a final checkout of the system. The particular requirements of each shakedown flight are dependent upon the subsequent data gathering exercise for which the aircraft is destined. Typically, for an aircraft

which has been allocated to a study of taileron, fin and rear fuselage loads, the list of manoeuvres to be performed during the shakedown flight is as shown on Table 6.

10. DATA COLLECTION

The seven SUMS aircraft will enter service over a two year period. At a peak approximately 3000 hours of data per year will be recorded. Data will be forwarded to BAe Warton on completion of every three flights per aircraft. The data package will consist of 3 flight cassettes, 3 identifying cassette proformas and associated fatigue meter readings, sortie code and stores information. The cassette data will be checked out on a ground replay unit and will then be transferred to a computer compatible tape (CCT). Identifying header information, including the fatigue meter readings and associated sortie data, will also be added to the CCT (Table 7).

11. GROUND ANALYSIS

A number of studies will be performed using the recorded data once it has been verified and synchronised (Fig. 5). These are described below.

1. Accelerometer responses will be used to calculate aircraft cg normal accelerations and the resulting time history will then be analysed using fatigue meter counting methods so that a comparison can be made with recorded fatigue meter data.

2. The "hot spot" strain histories will be analysed by rainflow techniques and the results compared with those from similar gauges on the airframe fatigue test.

3. Following a conversion of the strain gauge responses into load-time histories load spectra for different structural components (fin, taileron etc.) will be derived as a data base for the future.

4. For a number of selected areas of the structure (chosen as being likely to be the most important from a fatigue point of view - see Table 8) stress-time histories will be generated using relationships similar to those outlined in section 8. Matrices of mean and alternating stresses will then be derived by means of a rainflow analysis using an RAE program (refs. 12 and 13) and fatigue damage per sortie code calculated. The fatigue performance data for each selected structural feature will be based on either major airframe fatigue test knowledge or generalised design criteria. Test experience will take precedence.

5. For areas monitored by the fatigue meter the damage per flight will be compared with that derived from the load-time history traces.

6. A limited study to determine the damaging areas within each sortie code will be performed as an aid to fleet management and fatigue conservation. This study will include the use of plots of cumulative damage versus time (13).

7. The damage/sortie data will be used for fleet management purposes.

8. In the longer term studies will be made of the use of aircraft parameters to monitor fatigue usage. These studies will attempt to form load predictor equations from measured load and parameter time histories (see section 3) with two objectives in mind.

(i) to enable the longer term monitoring on the seven SUMS aircraft to be accomplished using parameter recording rather than by using strain gauge responses (a greater degree of long term integrity was expected from the parameter transducers)

(ii) to enable studies to be made of simplified methods of monitoring for possible fleetwide usage (for example can fin load be monitored simply by measuring rear fuselage lateral acceleration?).

12. CONCLUSIONS

Following a summary of the background work which led to the definition of the requirement for a comprehensive fatigue monitoring study on seven RAF Tornado aircraft a brief description of the instrumentation, calibration and ground analysis techniques was discussed.

Initially the main effort was seen to be based on the use of loads data derived from strain gauge responses from fully load calibrated aircraft. At a later stage it is planned to make use of load prediction equations derived from flight response parameter data.

Longer term plans to examine simplified methods of approach for fleetwide fatigue monitoring are also noted.

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ACKNOWLEDGEMENTS

The development of a system such as SUMS involves many people in industry, at the research establishments and in the Royal Air Force. Special acknowledgement is made of the major contributions to the programme by Dorothy Holford and John Sturgeon (RAE Farnborough), Wg Cdr D M F Bright (MoD RAF) and Stuart Lewis and Paul Titcombe (BAe). In particular the studies into the use of the parametric approach to load prediction undertaken by Stan Billington, a BAe colleague who was tragically killed in a gliding accident a year ago, are gratefully acknowledged.

TABLE 1

DEVELOPMENT AIRCRAFT "LOADS" CHANNELS

ITEM		CHANNELS
Starboard wing	- main surface	9
	- slat tracks	6
	- slat jacks	2
	- spoiler position	2
Port wing	- main surface	6
	- flap tracks	5
	- spoiler jacks	2
	- flap jacks	2
	- Kruger flap jack	1
	- spoiler position	2
Starboard taileron		5
Port taileron		5
Fin		7
Heat exchanger fairing		2
Rear fuselage		13
Front fuselage		2
Rudder jack		1
Airbrake jack (port)		1
Starboard wing sweep actuator		1
Port wing sweep actuator		1
Total load channels, excluding stores		75

TABLE 2

SELECTED DATA RECORDING FOR FIRST AIRCRAFT

ITEM	SAMPLES/SECOND
15 stain gauge responses	16 each
1 strain gauge response	32
elapsed time (minutes, seconds)	2 + 2
airspeed	4
altitude	4
outside air temperature	4
flap position	4
slat position	4
H P rpm (port)	4
H P rpm (starboard)	4
wing sweep angle	2
power lever setting (port)	2
power lever setting (starboard)	2
nozzle area (port)	4
nozzle area (starboard	4
taileron position (port)	16
taileron position (starboard)	16
rudder position	8
outboard spoiler (port)	16
inboard spoiler (starboard)	16
6 accelerometers (3z, 2x, 1y)	16 each
Total	486

The remaining 26 samples/second are used for control purposes.

During quiescent period all sampling rates are compressed to 1 sample/second.

TABLE 3

STRAIN GAUGE CHANNELS TO BE RECORDED ON 1st AIRCRAFT

- Wing root bending, port
- Wing root bending, starboard
- Wing station 1 bending, starboard
- Wing station 2 bending, starboard
- Rear fuselage bending
- Taileron root shear, port
- Taileron root bending moment, port
- Taileron root torsion, port
- Taileron jack load, port
- Taileron root shear, starboard
- Taileron root bending moment, starboard
- Taileron root torsion, starboard
- Taileron jack load, starboard
- Fin root shear
- Fin root bending moment
- Fin root torque

TABLE 4

CALIBRATION LOADING - SUMMARY

Wing loading points	10	
Loading cases, uncombined strain gauge bridges		16
Loading cases, combined strain gauge bridges		21
Fin loading points	11	
Loading cases, uncombined strain gauge bridges		21
Loading cases, combined strain gauge bridges		21
Taileron outer bearing cases	8	
Taileron panel loading points	9	
Loading cases		20

TABLE 5

ACCURACY OF THE CALCULATION OF LOCAL LOADS OR STRESSES FROM OVERALL LOADING

STRUCTURAL COMPONENT	PARAMETER	LOCATION	FORM OF EQUATION	CORRELATION
Taileron Skins	spanwise stress	rear spar, outboard	1	1,0000
	spanwise stress	rear spar, inboard	1	1,0000
	chordwise stress	root rib, forward	1	0,9999
	spanwise stress	front spar, outboard	1	1,0000
Fin	end load	forward pick up	1	0,9982
	end load	centre pick up	1	0,9999
	end load	rear pick up	2	1,0000
	principal stress	front spar	1	0,9916
Wing Skins	principal stress	inboard pylon	8	0,9964
	principal stress	outboard pylon	4	0,9999
	principal stress	inboard rib	3	1,0000
Wing Carry-Through Box	end load	aft, outboard link	5	0,9999
	end load	forward, outboard link	6	1,0000
	end load	aft, centre post	6	0,9714
	end load	forward centre post	7	0,9997

NOTES

Equations are of the form:

1. Parameter = $aM + bS = cT$
2. Parameter = $aM + bT$
3. Parameter = $aM1 + bM2 + cT$
4. Parameter = $aM + bT1 + cT2$
5. Parameter = $aS + b(M\sin\theta + T\cos\theta)$
6. Parameter = $aM1\cos\theta + b(M2\sin\theta + T\cos\theta)$
7. Parameter = $aS + bM1\cos\theta + c(M2\sin\theta + T\cos\theta)$
8. Parameter = aM

where a, b, c are coefficients

M, T, S are SUMS measured bending moments, shear torque

θ is a measure of wing sweep

TABLE 6

DEDICATED SHAKEDOWN FLIGHT MANOEUVRES FOR AIRCRAFT ALLOCATED TO EMPENNAGE LOADING STUDIES

Prior to flight

1. Record all ground datums
2. Move all control surfaces their full travel and hold

During flight

1. Perform datum manoeuvres
   - 1.1 0 to 2g roller coaster at forward wing sweep
   - 1.2 Steady heading sideslip
2. Forward wing sweep
   - 2.1 Wind up turn
   - 2.2 Three rapid rolls at different 'g'/M/alt combinations
3. Mid wing sweep
   - 3.1 Wind up turn
   - 3.2 Six rapid rolls at different g/M/altitude combinations
4. Aft wing sweep
   - 4.1 Wind up turn
   - 4.2 Five rapid rolls at different g/M/altitude combinations
5. Repeat datum manoeuvres before landing

Post flight

1. Record all ground datum values.

TABLE 7

HEADER BLOCK INFORMATION ON THE COMPUTER COMPATIBLE TAPE

Cassette serial number and flight number
Date recorded and replayed
Aircraft Tail Number
Sortie pattern code
Fatigue meter counts
Fuel at start
Fuel received (air-to-air)
Fuel at shut down
Stores at each of seven pylon locations

TABLE 8

FATIGUE MONITORING LOCATIONS

GENERAL AREA	NUMBER OF LOCATIONS
Forward fuselage	1
Centre fuselage	7 + 7
Rear fuselage	6 + 3
Wing	3 + 3
Wing carry through box	5 + 3
Taileron	6 + 6
Fin	4

NOTE: Locations shown a + b imply monitoring on both port and starboard sides

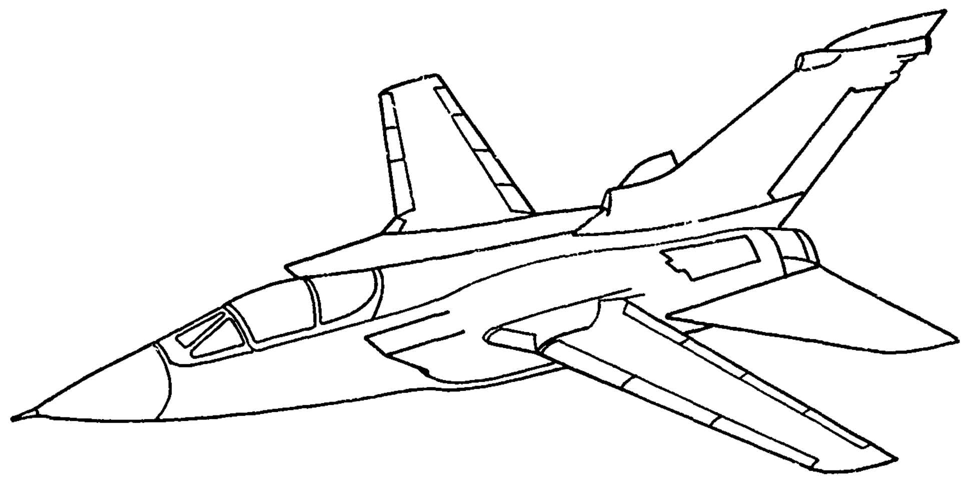
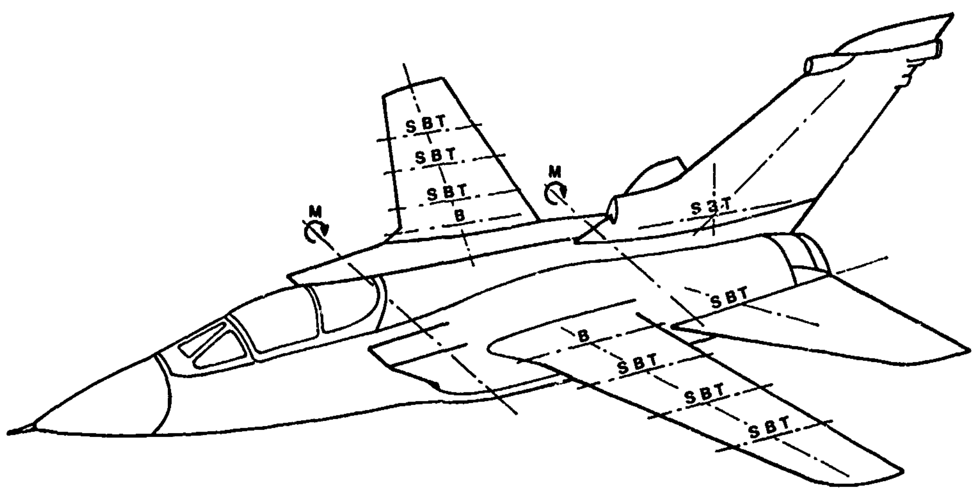


Fig1 TORNADO



S Shear
B Bending Moment
T Torsion
M Fuselage Vertical Bending

Fig2 Load Monitoring Stations

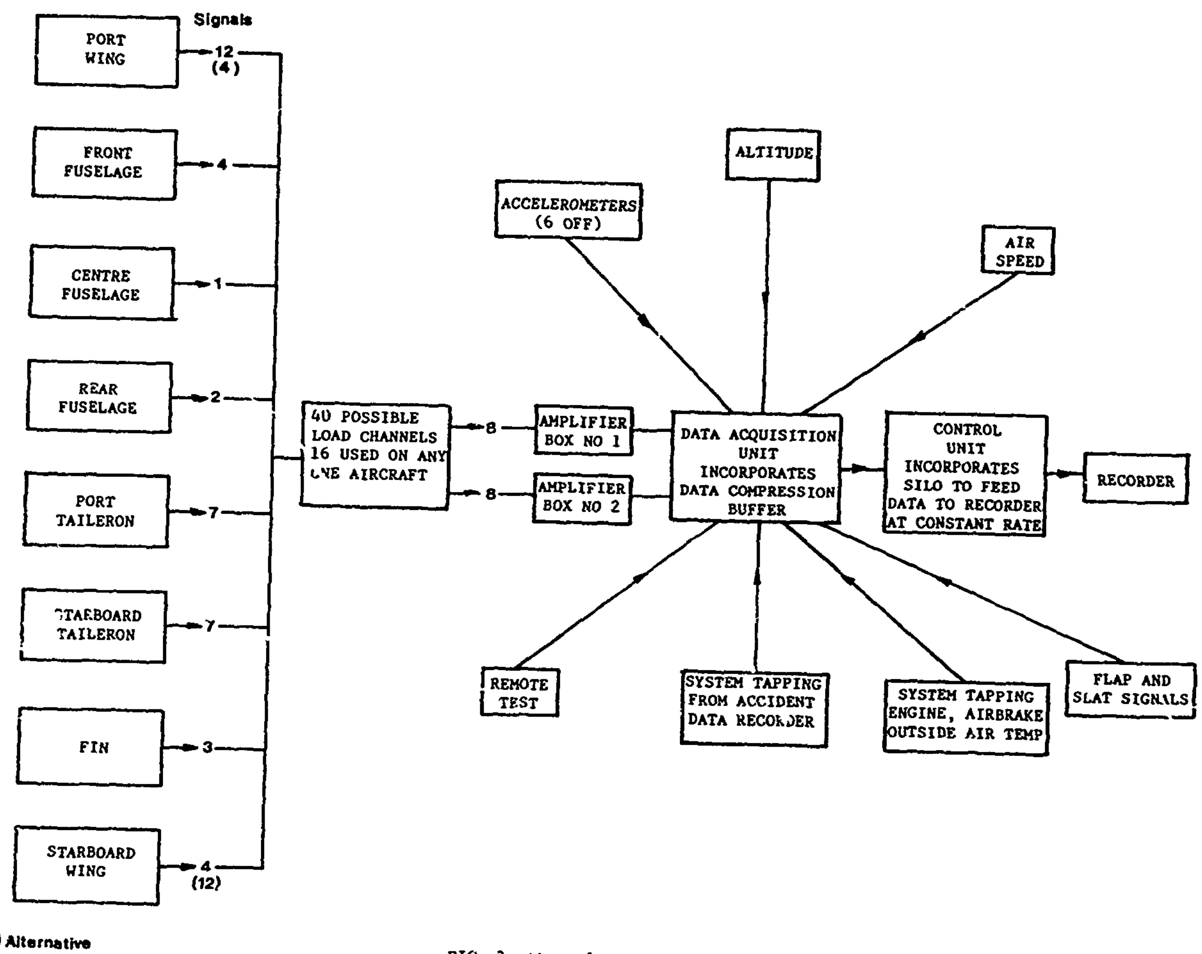


FIG. 3 Aircraft Instrumentation Block Diagram

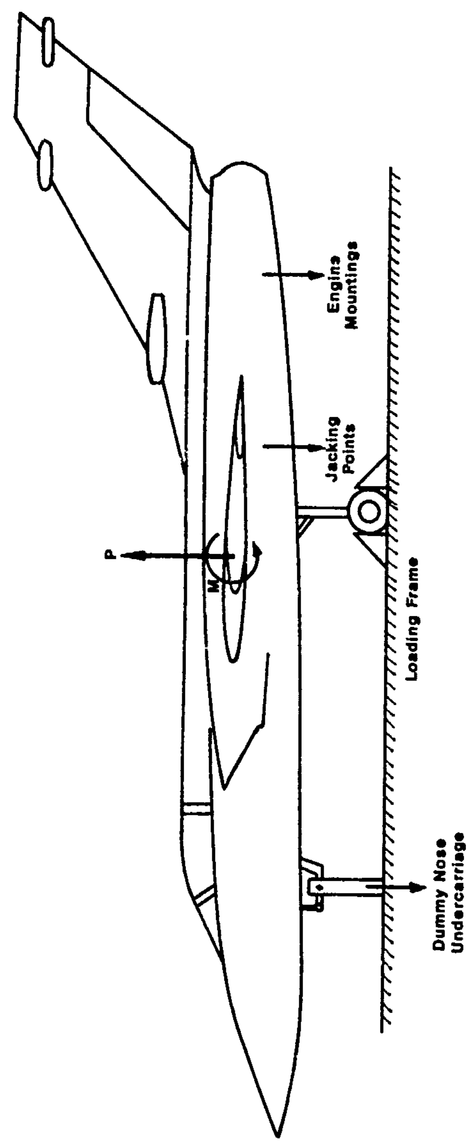


Fig4 Reactions for Wing Load Calibration

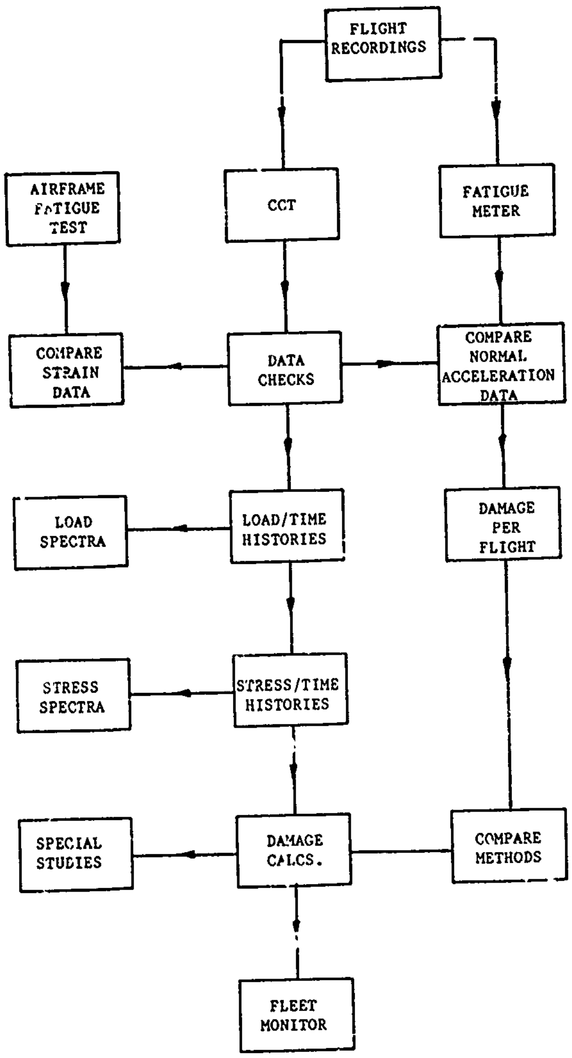


FIG. 5 Simplified Analysis Flow Chart

SUIVI DES CHARGES EN SERVICE DANS L'ARMEE DE L'AIR FRANCAISE
DEFINITION D'UN NOUVEAU MATERIEL

L. BARANES ET J.P. CORNAND.STPA 4, Av. de la Porte d'Issy PARIS (FRANCE)

I.- INTRODUCTION.

Cet exposé présente brièvement le développement d'un nouveau système de suivi des charges en service pour les avions de l'Armée de l'Air Française. Ce système a été développé pour remédier aux lacunes des systèmes actuels (accélérocompteurs) en offrant une possibilité d'acquisition de paramètres multiples associée à une meilleure logique de réduction de données. Les points suivants sont présentés .

- Le suivi des charges actuel, en insistant sur les problèmes posés par l'utilisation de "simples" accélérocompteurs.
- La définition des caractéristiques d'un nouveau système et l'expérimentation des prototypes.
- La description de la version de série de ce nouveau système (SPECS).
- Les applications envisagées et les développements futurs considérés pour le système.

II.- HISTORIQUE - SITUATION ACTUELLE.

A l'heure actuelle la plupart des flottes d'avions de combat de l'Armée de l'Air sont équipés dans une proportion importante d'accélérocompteurs comme moyen standard de suivi des charges en service. (Cf. Pl. 1)

L'utilisation de ces appareils est maintenant généralement acceptée et a déjà permis de montrer l'utilité du suivi des charges sur les avions de combat.

Sans revenir sur les différents avantages qu'on peut théoriquement tirer de ce suivi, on peut néanmoins citer deux exemples de retombées positives :

- Le Mirage F1 emporte en bout de voilure deux engins Magic. Ces engins sont munis d'ailettes importantes pour permettre une bonne manoeuvrabilité, mais quand ils sont liés à l'avion, ils subissent de ce fait des efforts aérodynamiques importants qui accroissent notablement le moment de flexion par G le long de la voilure. Combiné avec le pourcentage important de vols où ces engins sont emportés, ce fait avait une influence négative très importante sur la tenue en fatigue de la voilure.

Cet effet a été mis en lumière par la campagne de suivi des charges sur le F1 et en conséquence il a été décidé d'utiliser pour les missions non opérationnelles un Magic d'entraînement à ailettes rognées, voire supprimées, afin d'annuler cette augmentation du moment de flexion.

- Le JAGUAR est un avion d'appui franco-britannique en service dans les Armées de l'Air des deux pays. Le suivi des charges dans l'Armée de l'Air française a montré une utilisation nettement moins sévère que dans la Royal Air Force et a ainsi permis de limiter et de retarder l'application de modifications coûteuses, avec les économies correspondantes.

D'un autre côté l'utilisation d'accélérocompteurs a donné lieu à un certain nombre de difficultés liées aux limitations de ce système :

. Appareil monocanal : c'est la limitation la plus évidente : ce système ne donne pas d'information sur un phénomène non lié directement au paramètre mesuré (dans ce cas l'accélération verticale au centre de gravité). Parmi les points d'intérêt non accessibles figurent par exemple les charges sur les parties arrières de l'avion ou les charges dissymétriques.

. Logique de comptage. La logique de réduction de données adoptée était facile à réaliser avec la technologie disponible quand cet appareil a été conçu, mais elle implique une importante perte d'information.

Cette perte peut conduire à des problèmes lorsqu'on doit reconstituer un spectre avec les quelques points de mesure obtenus (nécessité d'établir des règles d'interpolation et d'extrapolation), surtout quand les niveaux de comptage ne sont pas bien adaptés au spectre à obtenir. Le résultat final (par exemple une vitesse d'endommagement en un point de la structure) est lui très sensible à la bonne adéquation de ces règles.

On peut citer comme exemple le cas d'un avion de surveillance maritime pour lequel les niveaux de comptage n'étaient initialement pas très resserrés dans la bande 1,5 à 2 G. Le spectre mesuré étant assez plat, il était considéré satisfaisant d'utiliser une interpolation linéaire entre deux niveaux de comptage voisins, en l'occurence 1,3 et 1,8 G. Cependant comme les charges de cette amplitude étaient les plus endommageantes - compte tenu de leur fréquence -, il a été décidé d'augmenter la précision des mesures dans cette zone (au détriment d'autres, le nombre total de niveaux de comptage étant fixé) en introduisant des comptages aux niveaux 1,3, 1,5, 1,65 et 1,8G. Cette introduction a permis l'établissement de règles d'interpolation légèrement différentes qui, appliquées à la même base de données, se traduisent par une différence de près de 20 % avec la durée de vie initiale. (Cf. Pl. 2)

De plus cette logique de comptage ne permet pas d'appréhender deux effets auxquels les modèles actuels de fatigue et de propagation accordent une place importante.:

- La séquence des charges

- La valeur des charges maximales subies : le nombre limité de compteurs disponibles oblige en effet à répartir les niveaux de comptage sur toute la plage des facteurs de charge utilisables avec une densité assez faible (typiquement 1 à 1,5G entre deux niveaux consécutifs pour un avion de combat).

. Recueil des informations : la nécessité de lire et transcrire manuellement les informations crée une charge de travail très raisonnable pour de petites campagnes de suivi, mais de plus en plus difficile à accepter pour le suivi à long terme de flottes importantes.

III.- NOUVELLE PERCEPTION DU BESOIN.

Ces dernières années ont vu se développer en parallèle deux phénomènes :

- la prise de conscience par l'utilisateur que le suivi en service pouvait avoir des effets concrets positifs. Bien que la mise en place de ce suivi ait commencé au milieu des années soixante dix, les résultats ne sont devenus vraiment significatifs et exploitables que lorsque la majorité des avions d'une flotte a été équipé pendant un pourcentage important de leur temps de vol. Jusqu'à ce moment là les informations fournies étaient de nature statistique, ce qui, sans manquer d'intérêt, offrait peu de possibilités d'action. Par contre une bonne connaissance du passé de chaque avion permet d'envisager des actions de gestion de flotte, et l'arrivée à ce stade a fait naître dans l'Armée de l'Air un intérêt croissant, associé à des demandes nouvelles (présentation des résultats, travaux d'extrapolation ...)

- la prise de conscience de plus en plus aigüe, au sein des organismes chargés de l'exploitation des mesures, des difficultés et limitations citées plus haut :

. appareil mono-canal,

. logique de réduction de données peu performante,

. transcription manuelle des données

et de la nécessité de disposer d'un moyen d'investigation plus performant, en particulier au vu des points suivants :

- le souci d'étudier des charges non liées au facteur de charge vertical (charges sur empennages horizontal ou vertical par exemple, charges liées à une utilisation particulière : tir canon, vrilles ... à une configuration particulière : charges sur dispositif hypersustentateurs).

- le souci d'étudier les charges subies par des avions plus gros où la souplesse de la structure à une influence importante sur les charges qu'elle subit (avions de transport par exemple).

- le souci d'identifier mieux les causes de l'endommagement des avions. Ce souci est en particulier motivé par la constatation faite ces dernières années selon laquelle la sévérité de l'utilisation des flottes allait en croissant, sans que cette tendance ait trouvé une exmplication rationnalle et certaine. (Cf. Pl. 3)

- la complication importante apportée au suivi des charges par l'utilisation de commandes de vol électriques sur des avions comme le Mirage 2000, qui introduisent des couplages dont les effets sont difficiles à cerner sans le suivi d'un ensemble complet de paramètres.

Pour illustrer ce point on peut indiquer que des charges majorantes sur la dérive du Mirage 2000 ont été obtenues dans un cas de ressource symétrique, avec emport de charges dissymétriques = la dissymétrie de la configuration entraîne un braquage des élevons dissymétriques et un braquage du drapeau, les CDVE cherchant à maintenir pendant la manoeuvre les ailes horizontales et l'absence de dérapage. Dans un cas de vol où cette manoeuvre atteignait la saturation des servocommandes des élevons, le braquage drapeau demandé s'est trouvé excessif et a induit un dérapage important.

La prise de conscience de ce besoin n'a bien sûr pas été limitée à la France et le développement de nouveaux systèmes d'acquisition a été lancé dans un certain nombre de pays. Cependant les matériels connus à la fin des années 70 ne paraissant pas répondre complètement aux besoins tels qu'ils étaient perçus en France, les Services Officiels Français se sont penchés sur la définition d'un nouveau type de matériel apte à compléter l'accéléro-compteur.

Ce travail a été pour sa plus grande partie effectué au CEAT (Centre d'Essais Aéronautique de Toulouse), fort bien placé pour l'effectuer puisqu'il a la charge de l'exploitation de pratiquement toutes les campagnes de relevés sur les avions militaires français.

La réflexion s'est articulée autour des axes suivants :

a) Le matériel à définir devrait pouvoir servir :

. à l'étude d'un problème identifié,

. à la bonne connaissance des charges sur quelques avions d'une flotte, afin de permettre un recalage des autres avions équipés d'un moyen de suivi plus simple (cette optique correspond sensiblement à celle exprimée par l'USAF dans l'ASIP (Air Force Structural Integrity Program) : MIL-STD-1530).

b) Il devrait fournir une information de bonne "qualité", c'est-à-dire permettant d'effectuer, immédiatement ou a postériori, des corrélations entre paramètres et des calculs de fatigue ou de mécanique de la rupture suivant toute loi de comportement choisie par l'utilisateur.

c) Il devrait être suffisamment petit pour être installé sans dégradation des performances sur un avion de combat opérationnel et son utilisation devrait poser la contrainte la plus faible possible au personnel mettant l'avion en oeuvre.

d) Il devrait être autant que possible banalisé pour que son adaptation d'une utilisation à l'autre soit aisée. Il a été considéré dès l'origine que ce matériel ne serait probablement installé qu'en petit nombre sur un type d'avion donné et pourrait être utilisé successivement pour plusieurs applications.

e) La gestion et l'exploitation des informations recueillies devrait être la plus simple possible.

Les principaux choix à faire portant sur les informations à recueillir et le principe de fonctionnement, ceux-ci ont été examinés successivement :

<u>Informations à recueillir.</u>

Elles sont de deux natures :

- les "paramètres d'utilisation" (date, unité, base, mission, configuration, évènements spécifiques au vol ...) nécessaires pour une exploitation statistique fine et dans certains cas pour le calcul des charges lui-même,

- les paramètres permettant d'accéder à l'évolution des charges aux points de l'avion considérés. Ils peuvent être en nombre très variable et de différents types :

  - jauges de contraintes,
  - paramètres avion : vitesses, accélérations, altitude, Mach, braquages, configuration aérodynamique, masse ...
  - temps.

<u>Principe de fonctionnement.</u>

Les "paramètres d'utilisation" cités ci-dessus sont discrets (une valeur par vol) et ne sont en général pas accessibles par une mesure mais doivent être obtenus auprès de l'utilisateur.

Le traitement des autres paramètres pour obtenir par exemple une estimation de l'endommagement peut se faire selon différents procédés indiqués dans la planche 4.

Les principales remarques qu'on peut faire sur ce tableau sont :

. les matériels des colonnes 1 et 2 sont trop rudimentaires ou pas suffisamment au point dans leur état actuel pour répondre au problème,

. les matériels de la colonne 3 nécessitent un volume considérable de stockage à bord et de traitement au sol, certainement rédhibitoire pour un enregistreur multiparamètres. Ils ne semblent guère adaptés au problème du suivi continu, sauf dans certains cas où des avantages spécifiques peuvent les rendre intéressants,

. le principe de la colonne 5 présente un inconvénient majeur : celui d'utiliser un calcul nécessairement figé et de ne pas laisser subsister les informations intermédiaires nécessaires à d'autres traitements (statistiques, corrélations, passage d'un modèle d'initiation à un modèle de propagation de fissure par exemple).

Le principe d'enregistrement avec réduction de données a donc été retenu, car compatible avec les objectifs recherchés. Restait à choisir la logique de réduction de données qui permettrait la perte d'information minimale.

Après examen des logiques les plus fréquemment rencontrées (Cf. planche 5) et au vu des critères de choix exposés ci-dessus, il a été décidé de retenir une logique associant la détection d'extrema et la notion de paramètre "pilote" :

. <u>Notion de paramètre pilote.</u>

Il a été considéré que si la reconstitution de l'histoire des contraintes en un point pouvait nécessiter la connaissance d'un nombre relativement élevé de paramètres (sauf évidemment si une mesure directe est faite en ce point), un certain nombre de ces paramètres étaient à variation relativement lente voire nulle (ex : masse de carburant, Mach, configuration ...) et donc que ce n'était pas ceux-ci qui détermineraient un évènement significatif. On a donc décidé d'isoler et de ne "traiter" par la logique de réduction de données que le petit nombre de paramètres à évolutions rapides (ex : facteur de charge, vitesse de roulis ..)

permettant d'isoler ces évènements significatifs . Lorsque la logique de tri aurait détecté un extremum (voir ci-dessous) sur un de ces paramètres, dits "pilotes" dans la suite, l'ensemble des autres paramètres serait acquis simultanément à cet instant jugé intéressant. (Cf. Pl.7)

. Détection d'extrema.

La logique de tri retenue est l'extraction et le stockage séquentiel des extrema. (Cf. Pl. 6)
Ce tri s'effectue à partir d'un algorithme simple, basé sur :

- la numérisation du signal à une cadence suffisante au vu des fréquences caractéristiques des phénomènes considérés.

- la reconnaissance d'un extremum

- la reconnaissance du caractère significatif d'un extremum, basée sur une valeur de "seuil" choisie a priori.

  (Un extremum est jugé significatif et donc conservé si il est séparé de l'extremum précédent, et donc de sens opposé, par une valeur au moins égale à ce seuil).

Cette logique présente les avantages suivants :

. elle permet l'utilisation de l'ensemble des lois de comportements actuelles ou envisagées,

. elle permet une bonne reconstitution de l'histoire d'un vol, y compris la chronologie des évènements, la distribution entre les différentes phases du vol ...,

Par contre elle présente quelques inconvénients :

. elle nécessite un volume de stockage relativement important,

. ce volume est variable d'un vol à l'autre, et de manière imprévisible, puisqu'il dépend de la sévérité du vol. Cette variabilité nécessite de conserver des marges au niveau du moyen de stockage utilisé pour ne pas risquer de perdre de l'information.

IV.- DEVELOPPEMENT DU SPEES.

Les objectifs et les principes dégagés ont conduit le CEAT à développer un système probatoire destiné à montrer la faisabilité de l'approche considérée.

Ce système, connu sous le sigle SPEES : Système Prototype pour l'Etude de l'Endommagement Structural, a été construit avec les performances suivantes :

- capacité d'acquisition de 16 paramètres dont 4 pilotes, plus le temps écoulé depuis le début du vol,

- cadence de numérisation de 50 Hz

- afin de ne pas être limité par la capacité de stockage pour les expérimentations, un système de stockage à capacité très large a été choisi, en l'occurence un enregistreur magnétique.

Ce matériel, construit sans satisfaire aux normes aéronautiques et sans souci immédiat de compacité, s'est révélé relativement volumineux :

calculateur : 1 ATR long
+ enregistreur : 1/4 ATR court.

Ce matériel a connu deux expériences d'utilisation :

- quelques vols sur Mirage III B du CEV, l'appareil étant utilisé au maximum de sa capacité. Cette opération a mis en évidence le bon fonctionnement de l'appareil mais le caractère limité de l'expérience n'a pas permis d'examiner l'ensemble des problèmes d'utilisation opérationnelle ;

- une campagne sur Transall en service, l'appareil étant uniquement relié à un accéléromètre dans une première phase. Cette campagne a fait apparaître une lacune importante : l'absence d'acquisition des param-tres d'utilisation qui doivent actuellement être recueillis séparément, en vue d'une adjonction aux mesures durant le traitement : Ce procédé est très contraignant et présente des risques d'erreurs importants.

Cependant cette campagne limitée a déjà permis d'illustrer de façon parfois spectaculaire la très grande diversité des charges subies par cet avion en fonction du profil de mission suivi. (Cf. Pl. 9)

A la suite de ces premiers résultats encourageants, il a été décidé de procéder au développement industriel du système. Après consultation ce développement a été confié à la société Electronique Serge Dassault, et le sigle SPEES a été conservé, mais en prenant la signification Système Pour l'Etude de l'Endommagement Structural, marquant donc la perte du caractère prototype de ce matériel.

En parallèle avec le développement du matériel de série, ESD a réalisé à partir d'un matériel d'essais en vol existant un prototype industriel qui fait actuellement ses premiers vols sur un des prototypes du Mirage 2000. Les mesures recueillies serviront de support au développement des programmes d'exploitation actuellement en cours. La planche 8 donne la liste des paramètres initialement retenus pour cette opération.

V.- DESCRIPTION DU SPEES DE PRODUCTION.

La version initiale du matériel a comme caractéristiques principales :

. boîtier 1/4 ATR court (57 x 190 x 380 mm) masse < 5 kg (Cf. Pl. 10)

. alimentation 400 Hz triphasé

. Capacité d'acquisition :

20 voies analogiques haut niveau (0 à + 5 V) et/ou bas niveau (± 16 mV) (cas par exemple de jauges de contraintes : le système est alors capable de fournir l'alimentation de ces jauges). Les informations sont numérisées sur 8 bits

et 16 tops 0/+ 28V dont 2 tops pilotes.

. base de temps incorporée ; capacité de comptage d'un top (débit carburant)

. scrutation de l'ensemble des paramètres à 50 Hz

. capacité de stockage : 2 000 blocs de données
(1 bloc = un ensemble de 20 mesures + 16 états binaires + temps + totalisateur
+ indication du paramètre pilote ayant déclenché l'acquisition
+ bits de contrôle
= 32 octets

sur cassette amovible de 64 Koctets de mémoire REPROM
(soit une autonomie d'environ 5 h en supposant une acquisition en moyenne toutes les 10 secondes).

. A la suite de l'expérience précédente, le système a été doté en face avant d'un panneau (visualisation + boutons poussoirs) permettant le contrôle en piste de la chaîne d'acquisition et l'insertion à la fin de l'enregistrement d'un vol d'un bloc de données contenant les paramètres d'utilisation selon un format préprogrammé.

. De plus, et parallèlement à la logique de tri déjà exposée, le système effectue systématiquement une acquisition à intervalle fixe (typiquement 30 s), afin de permettre une bonne reconstitution de la mission effectuée, même dans les périodes où il ne se passe pas d'évènement jugé significatif.

Développements futurs envisagés.

L'architecture du système permet d'incorporer ultérieurement des options telles que :

- pour utilisation sur un avion disposant d'un bus de données (norme Digibus GAM-T-101), possibilité de couplage à cette ligne pour le recueil direct de paramètres disponibles sur la ligne.

- Pour un avion où le boîtier ne pourrait pas être placé dans un endroit facilement accessible, possibilité de déport du panneau de dialogue et du logement de la cassette.

- Pour l'étude des charges au sol (impact en particulier), possibilité d'incorporer un buffer permettant une scrutation à cadence élevée (500 à 1000 Hz) suivie d'un tri en léger différé.

VI.- UTILISATIONS PREVUES.

Deux types principaux d'utilisations sont envisagés :

. l'équipement d'avions pour répondre à un problème ponctuel identifié. Chronologiquement c'est ce type d'utilisation qui arrivera en premier, en particulier pour les avions suivants :

- JAGUAR : étude des charges latérales (sur les avions français pour comparaison avec les mesures effectuées au Royaume Uni).

- ALPHAJET : étude des charges sur parties arrières et sur extrémités de voilure, afin de déterminer les charges à appliquer dans un essai de fatigue de ces parties (l'essai de fatigue d'ensemble n'ayant impliqué que la partie centrale du fuselage et de la voilure).

. l'équipement d'un petit pourcentage d'avions d'une flotte donnée. Ce type d'utilisation répond aux soucis explicités plus haut et devrait donc permettre :

- d'acquérir une bien meilleure connaissance de l'utilisation des avions et des causes de leur endommagement,

- de recaler les mesures effectuées sur des avions équipés seulement d'accélérocompteurs,
- d'être capable de réagir sans délai à l'apparition d'un problème imprévu (découverte d'un dommage en service par exemple) si l'équipement en question est suffisamment complet.

- Le type de mesures et le mode d'exploitation choisis dépendront à la fois de l'avion support et de l'utilisation telle qu'elle est définie ci-dessus. Schématiquement les deux situations extrêmes envisagées sont les suivantes :

  . étude d'un problème ponctuel sur un avion à l'avionique simple :

  Les paramètres recueillis seront en majorité les mesures de jauges de contraintes, associées à quelques paramètres avion pour cerner la provenance des variations de contraintes mesurées. L'exploitation se fera selon des méthodes proches des méthodes classiques utilisées actuellement en essais en vol : établissement de spectres, passage aux efforts généraux après calibration (méthode de SKOPINSKI par exemple), établissement de corrélations entre paramètres ...

  . Suivi d'un avion à l'avionique évoluée :

  Sur ce type d'avion la priorité sera mise sur l'acquisition de paramètres avion (vitesses, accélérations ...) qui devraient être accessibles sans nécessiter de capteurs supplémentaires (sur une ligne bus ou directement sur les équipements). Ce type d'équipement semble à la fois mieux adapté aux objectifs de ce suivi a priori très général et plus facile (pour l'installation comme pour la maintenance) qu'un équipement en jauges. L'exploitation des mesures pour retrouver des contraintes (ou des efforts généraux) se fera à l'aide d'une méthode d'identification comme celle présentée par les AMD-BA dans un autre exposé de cette conférence.

  Il est cependant certain que l'établissement de la fonction de transfert entre les mesures faites et les contraintes cherchées sera beaucoup plus complexe que lorsqu'on utilisait un appareil monocanal comme l'accélérocompteur.

VII.- CONCLUSION.

Le système qui vient d'être brièvement présenté doit selon nous répondre aux besoins identifiés en France pour un meilleur suivi en service des avions de l'Armée de l'Air. Après une phase de développement et d'expérimentation initiale, il entre maintenant en utilisation effective, où ses capacités pourront être pleinement démontrées.

Pour l'avenir, nous pensons que les principes qui ont donné naissance à cet appareil doivent être développés dans deux directions :

- un appareil appliquant la même logique de tri à un nombre restreint de paramètres (1 à 4 par exemple), capable de remplacer avec un encombrement et un coût réduit l'actuel accélérocompteur.

- la disparition d'appareils de suivi des charges pour les avions futurs !

L'acquisition, le tri et le stockage de paramètres permettant de reconstituer et d'expliquer l'histoire des charges en un point de l'avion sera toujours - et de plus en plus - d'actualité sur ces avions, en particulier les avions de combat qui deviennent de plus en plus manoeuvrants et donc de plus en plus sensibles à la fatigue.

Par contre l'architecture de leur avionique fera que l'ensemble de ces paramètres sera certainement facilement accessible sur un bus de données. Il sera alors beaucoup plus simple d'intégrer à un des calculateurs de bord cette fonction "surveillance des charges" que d'introduire un nouvel appareil spécifique.

Cette intégration - que subiront probablement d'autres fonctions comme la surveillance des paramètres moteurs - suivra donc la voie ouverte par l'electronique de bord pour lesquels l'autosurveillance et le diagnostic des pannes sont désormais la règle.

Néanmoins cette intégration ne se fera que pour la prochaine génération d'avions et n'est guère envisageable pour les avions actuels, y compris les plus récents comme le Mirage 2000 qui entame à peine se vie opérationnelle. Un appareil comme le SPEES a donc encore une longue carrière devant lui.

PLANCHE 1 : ARMEE DE L'AIR FRANCAISE - EQUIPEMENT EN ACCELEROCOMPTEURS DES FLOTTES.

FLOTTE	POURCENTAGE D'APPAREILS EQUIPES	
MIRAGE 2000	MISE EN PLACE EN COURS	
MIRAGE F 1	100 %	
MIRAGE III + 5	10 %	(*)
JAGUAR	→ 100 % EN 1984	
MIRAGE IV A	40 %	
ALPHA-JET	100 %	
FOUGA	10 %	(*)
EPSILON	MISE EN PLACE EN COURS	

(*) CAMPAGNES STATISTIQUES SEULEMENT (PAS DE SUIVI INDIVIDUEL).

PLANCHE 2 : EXEMPLE DE PROBLEME D'INTERPOLATION DES SPECTRES.

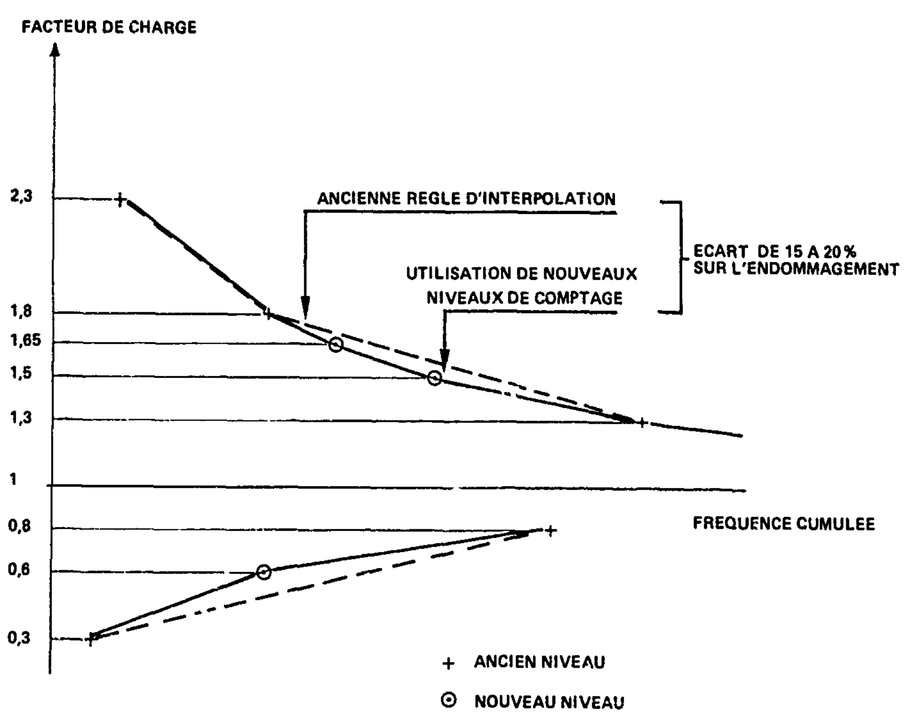


PLANCHE 3 : EVOLUTION DANS LE TEMPS DE LA SEVERITE DES SPECTRES (MIRAGE F1).

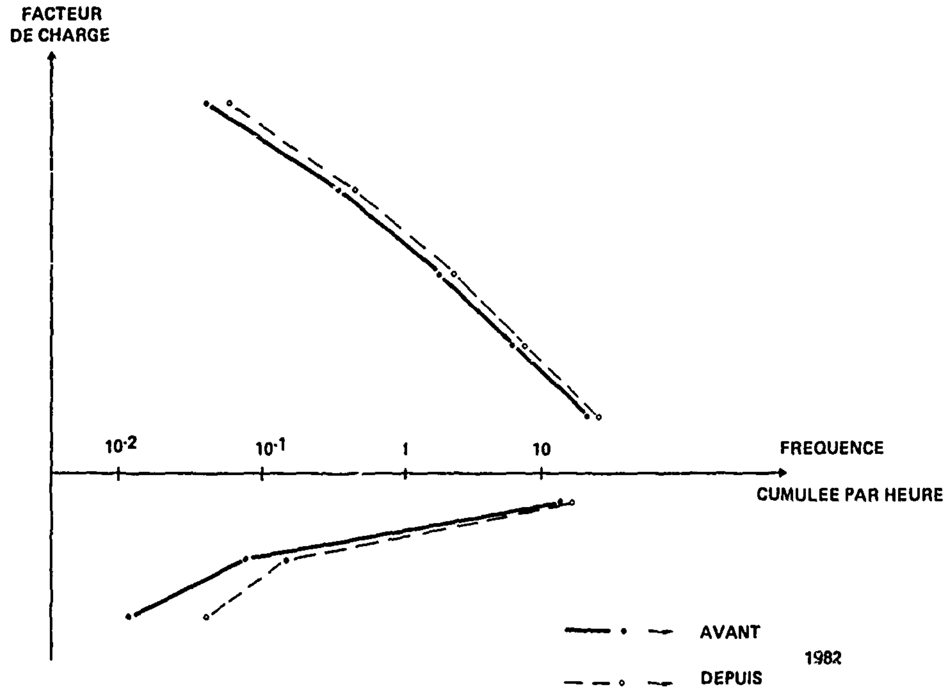


PLANCHE 4 : PRINCIPES DE FONCTIONNEMENT POSSIBLES.

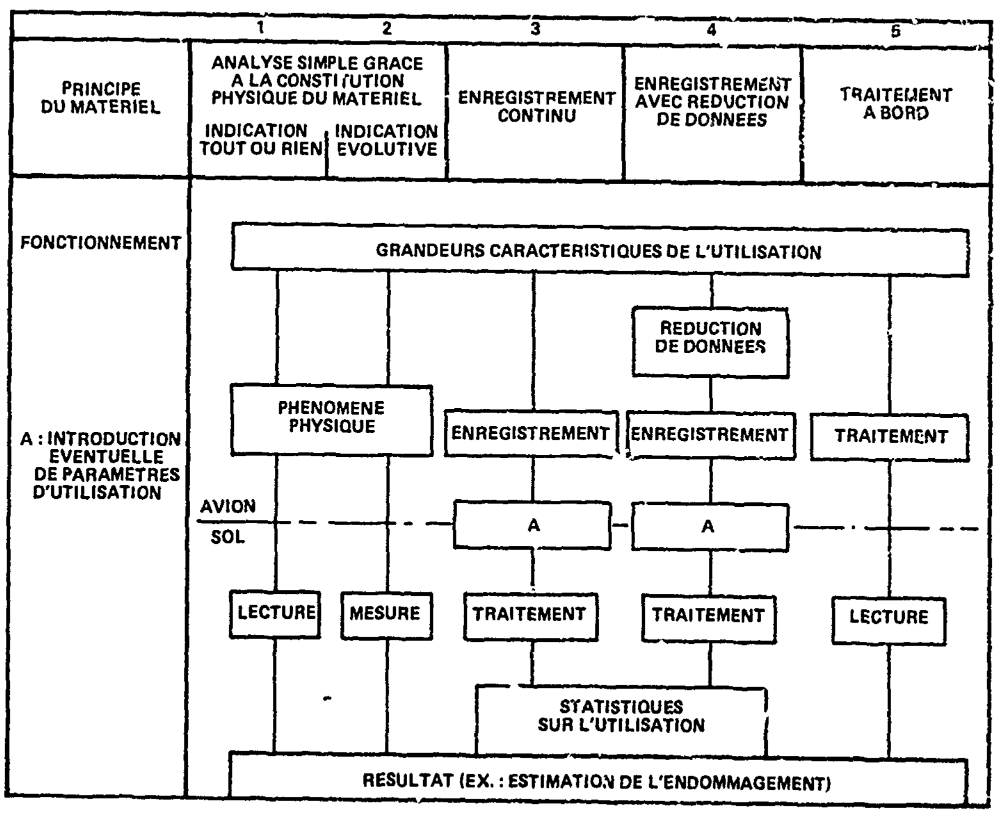


PLANCHE 5 : LOGIQUES POSSIBLES DE REDUCTION DE DONNEES.

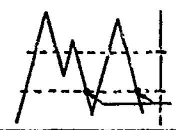
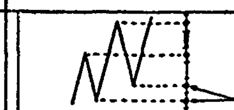
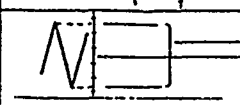
LOGIQUE	PRINCIPE SOMMAIRE	RESULTAT : TABLEAU
COMPTAGE CONDITIONNEL DE DEPASSEMENTS	 SEUIL DE COMPTAGE SEUIL DE REARMEMENT COMPTAGE	1 DIMENSION
EXTRACTION D'EXTREMA STOCKAGE SEQUENTIEL	SPEES	1 DIMENSION LONGUEUR VARIABLE
COMPTAGE DE PICS	 COMPTAGE	1 DIMENSION
COMPTAGE D'AMPLITUDE	 A1 A2	1 DIMENSION
COMPTAGE MOYENNE-AMPLITUDE	 AMPLITUDE MOYENNE	2 DIMENSIONS
"RAINFLOW"	 EXTRACTION SUCCESSIVE DE CYCLES IMBRIQUES	– SANS ORDRE DE CYCLE 2 DIMENSIONS – AVEC ORDRE DE CYCLE 1 DIMENSION x 2 LONGUEUR VARIABLE

PLANCHE 6 : LOGIQUE CHOISIE.
EXTRACTION ET STOCKAGE
SEQUENTIEL D'EXTREMA.

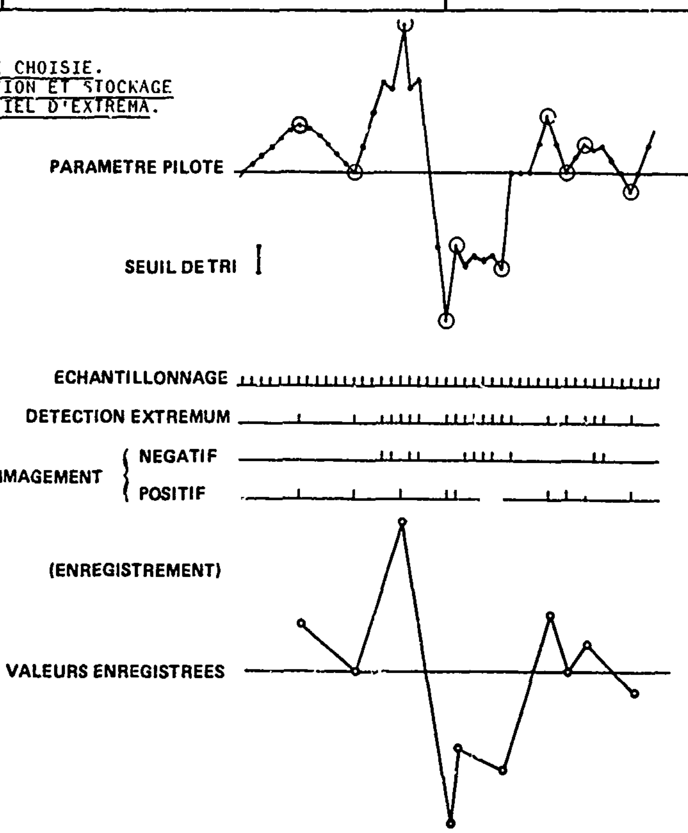


PLANCHE 7 : NOTION DE PARAMETRE PILOTE - ACQUISITION ET STOCKAGE PROVISOIRE.

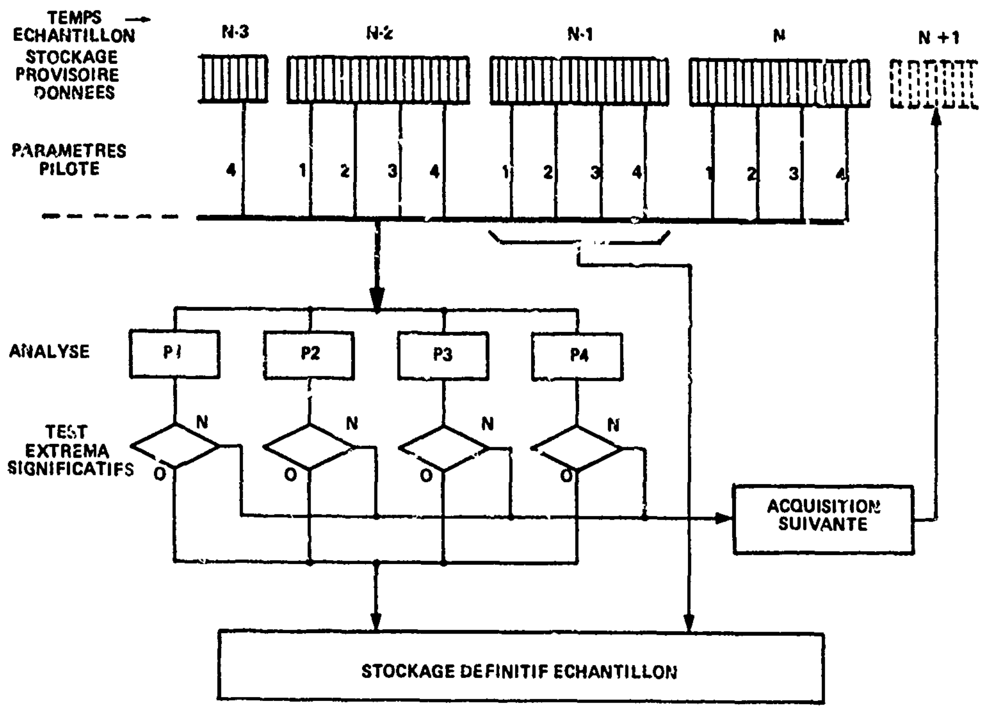


PLANCHE 8 : MIRAGE 2000-02 - LISTE DES PARAMETRES.

- VITESSE
- MACH
- PETROLE RESTANT
- FACTEUR DE CHARGE n_Z
- VITESSE DE ROULIS
- ACCELERATION DE ROULIS
- ACCELERATION DE TANGAGE
- INCIDENCE
- DERAPAGE
- BRAQUAGE ELEVON GAUCHE
- BRAQUAGE ELEVON DROIT
- BRAQUAGE DRAPEAU
- FLEXION VOILURE AU CADRE PRINCIPAL AV
- FLEXION VOILURE AU CADRE PRINCIPAL AR
- FLEXION DERIVE A ATTACHE AR
- POUSSEE MOTEUR

PARAMETRES PILOTES

PLANCHE 9 : SPEES PROTOTYPE : EXEMPLE DE MESURES SUR C-160 TRANSALL.

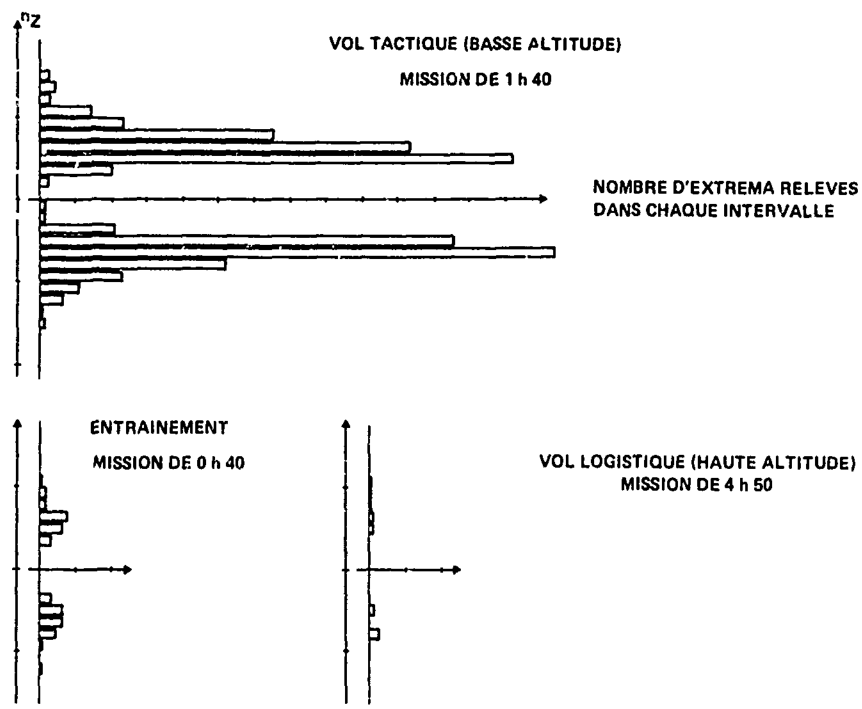


PLANCHE 10 : SPEES - PRODUCTION SERIE.

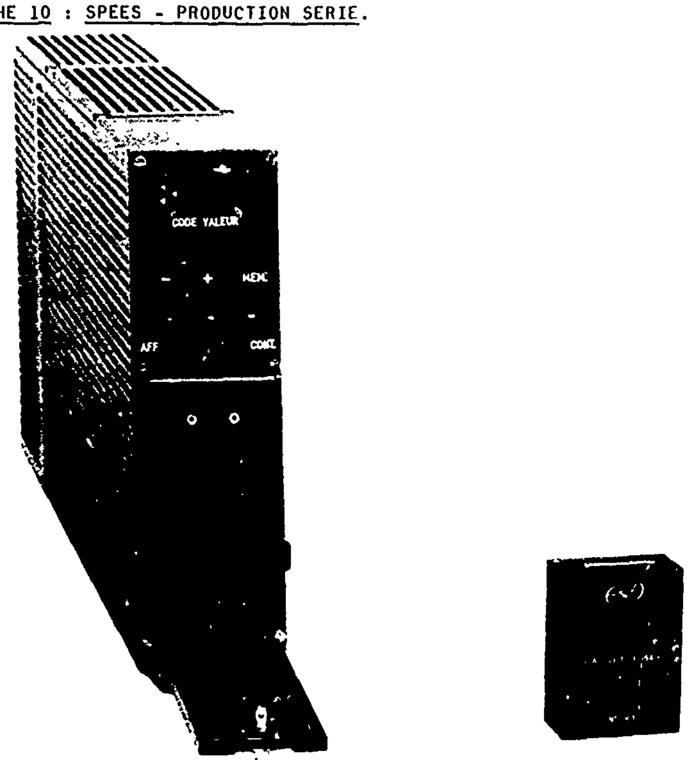
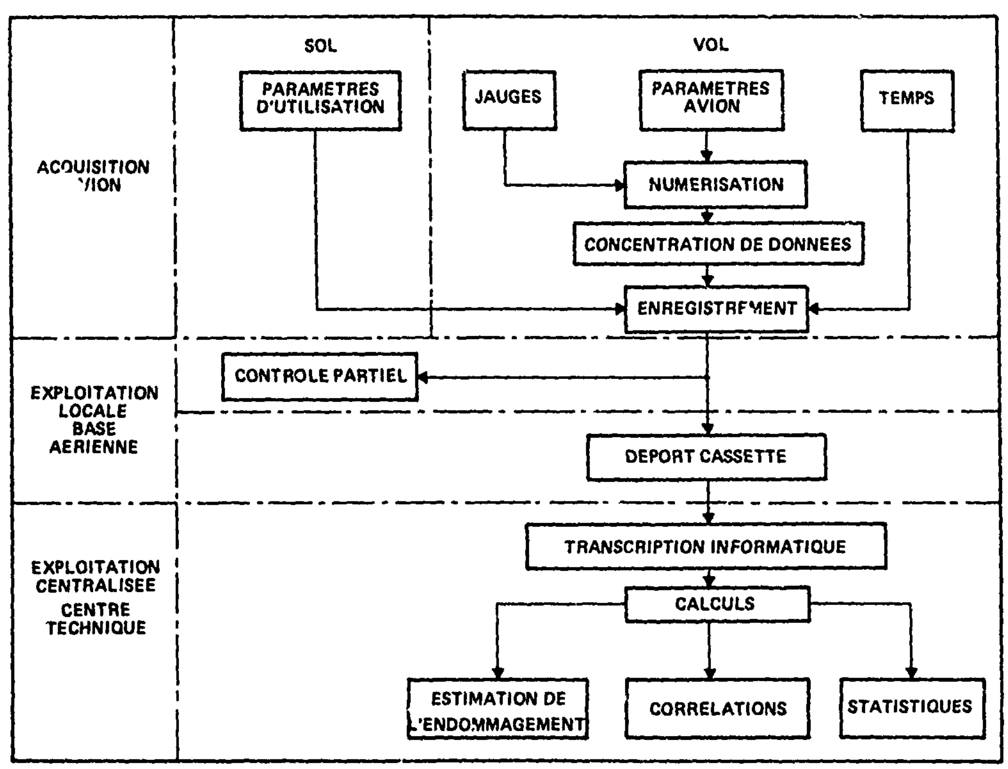


PLANCHE 11 : CONDITIONS D'UTILISATION ET D'EXPLOITATION.



FLIGHT PARAMETERS RECORDING FOR STRUCTURE FATIGUE LIFE MONITORING

by
F. Staropoli
AERITALIA
Gruppo Velivoli da Combattimento
Corso Marche, 41 - 10146 Torino
ITALY

SUMMARY

The philosophy shown in this work is based on the in flight parameters recording technique which becomes more inviting because of the trend of the aircraft to an ever increasing advanced technology and an ever increasing sophisticated avionic system for control and maneuverability purposes.
So, in these cases it is already available on board a large data amount which, in terms of parameters, can be recorded during flights on a magnetic tape in digital form.
It is avoided, by this way, the installation of several and various sensors (accelerometers, angular deflections transducers, etc.), being all the necessary parameteric informations directly extracted from the main computer without significant changes in the aircraft avionics.
The analysis and calculation have to be done on ground, where the computation capability can be convenient to meet the proposed aims.
Italian Air Force committed a job for the TORNADO Aicraft, based on the above philosophy, which provides, in addition to the structure fatigue life monitoring, the engines low cicle fatigue counting and full maintainability.

1. PARAMETERS RECORDING

The TORNADO maintenance data recorder system can operate on board and on ground in two different phases:

a) on board, reading all the relevant parameters from the Data Acquisition Unit and generally from the units which contain them.
A list of flight parameters, necessary to account for the fatigue life of each primary structural component, is shown in fig. 1. It has to be considered redundant for military aircraft with technologically low content and for civil aicraft which have simple flight profiles, while becomes necessary for the TORNADO because of the very large variety of missions, configurations and manoeuvre capability.
The sampling rate may be largely variable, depending on the kind of parameter and aircraft class: transport aircraft need a sampling frequence sensibly lower than fighters. However, 8 samples per second, as far as sampling rate is concerned, can satisfy the current metodology.

b) on ground, providing to read the recorded magnetic tape (cassette) in the Automatic Ground Station (AGS) . to analyze and elaborate the flight.
The AGS consists of a computer unit inproved by a disk and magnetic tape memory with graphical output facilities on printer - plotter.
The software has two block - programs sequentially arranged: the first one produces the external load time - histories on the monitored structural components; the results of this elaboration don't appear externally but they are memorized in a disk in order to be used by the second one which performs the structural damage calculations showing the whole output to the system's users.

2. EXTERNAL LOAD TIME - HISTORIES

A preprocessor in the AGS automatically prepares the data set which will be given to the computer program that selects the interesting slices, as far as structural fatigue damage is concerned, on the basis of the parameters variability, in order to save processing time.

To meet this aim, the recorded intervals to be analyzed are selected only when the most important parameters (vertical acceleration , rolling velocity, rolling acceleration, etc.) show significant magnitude variation. The remaining intervals are not analyzed.
The step subsequent to the choice of the interesting intervals, is the translation of the parameters time-histories into external loads historics on each relevant component.
To do it, two different ways were taken into account:

a) to use theoretical algorithms only.

b) to organize a normalized data bank obtained by external loading in flight recording procedure on prototype aircraft, in order to read by interpolation the in service manoeuvres, having the external loads output on the component.

The second approach, that allows a higher load calculation accuracy has to be preferred mainly when the aircraft is well known as far as flight test measurements are concerned.
This is the situation of the TORNADO which has a very large data bank, coming from several years of prototypes flight activity.
The fig. 2 shows the flow of the loads calculation steps.

3. STRUCTURAL FATIGUE DAMAGE COMPUTATION

The fig. 3 represents the calculation flow - chart of the fatigue analysis block program. The following paragraphs explain in a more detailed way the different aspects of the above analysis.

3.1 Main program and aircraft data bank

The main program takes two different inputs:

a) - from the cassette containing the recorded flight data

- Aircraft identification code
- Endurance (hours) of the flight (or flights) recorded on the cassette

b) - from the structural fatigue data bank (for each aircraft) :

- cumulated flight hours
- load - stress correlation factors to translate the load time - histories into stress - histories for each structural component.
  These factors are depending on the structural geometry and materials of each critical area and in addition they can take into account the differences, among the aircraft of the fleet, coming from several configurations (repairs, components substitution, redesign, etc.) .
- Cumulative damage values for the components.
- Safe fatigue life of the components.

The safe fatigue life of each component, stored in the fatigue data bank, is derived from test results.
In other words in the data bank the component tested fatigue lives only and not the design ones are stored, in order to perform a relative Miner rule calculation.
The data of an aircraft are retrieved from the data bank using the aircraft identification number written on the cassette.
The subsequent computation proceedes up to the last component and then two subroutines update the damage values in the aircraft data bank.

3.2 Data filtering and elaboration

The structural fatigue program has a subroutine using, as input data, the load time - histories.
The load values are transformed into stress values (see 4.1.b) by some conversion factors which are to be considered a proper characteristic of each individual aircraft.
The reliable acquisition of the above conversion factors is made from the stress analysis and the fatigue certification (fatigue tests with strain gauges measurements) .
Each stress value is compared with the previous one and rejected if no change in sign is discovered in the time - history derivative.
Using this way, a sequence composed by relative max and min values can be extracted.
In order to avoid the analysis of a large quantity of data, a filter rejects the stress peak values lower than $\Delta\sigma$ relative to the last accepted σ value.
The adopted $\Delta\sigma$ interval is equal to 5 N/mm^2 which is corresponding, for a cycle at R = -1 ($\sigma_{MAX} = 0.5\ N/mm^2$ and $\sigma_{MIN} = 0.5\ N/mm^2$) and 2024 - T3 aluminum alloy at KT = 3, to a damage value order of 1.E - 18.
The fig. 4 graphically shows the filter action.

3.3 Cycles counting method

A subroutine counts the stress fatigue cycles using the Rainflow Method of cycles definition.
It is widely known that this method allows to process a great number of data with a better cpu utilization.
Once a fatigue cycle is counted it is passed to the next subroutine for the damage calculation.

3.4 Damage calculation

The subsequent subroutine makes the components cumulated damage calculation.
It is made cycle by cycle following the Palmgreen - Miner rule which is considering, as consequence of a fatigue cycle, a life consumption equal to 1/N, where N is the number of cycles which is consuming the total life.
The fatigue curves are contained themselves in the structural fatigue program data bank, and they are strictly correlated with the test results.
The cycles wholly in compressive field are neglected by no damage attribution.
The cycles at "R" smaller than -1 are considered as combination of one cycle at R = -1 between σ_{MAX} and $-\sigma_{MAX}$, and an other one between $-\sigma_{MAX}$ and σ_{MIN} not damaging the structure (see fig. 5).
It has to be considered true below the material yielding thresholds.

3.5 Output organization

At the end of each damage calculation on the monitored components a subroutine provides for the updating of the aircraft data bank.
The fig. 6 shows the data bank organization.
It may be noted the presence of a dummy component (n. 21) that has to be considered as an ideal one which, loaded by the wing bending moment spectrum, allows a comparison, on a damage basis, of the aircraft of the monitored fleet.
From the above data bank a full output can be shown if requested from the AGS. It is arranged as reported in fig. 7 where the "equivalent flight hours" are obtained multiplying the safe fatigue life by the damage.
The standard AGS output, on the contrary, is a synthetic one organized and presented as in fig. 8, which is giving information referred to the component with the smallest residual life only.
Some other outputs are foreseen in order to find errors or to follow separately some elaboration phases, while some utility programs will provide changes in the data bank when and if some structural configuration modifications, concerning the monitored elements, will occur.

4. CONCLUSIONS

The system complexity is clearly pointed out.
However it is relaxed by the very easy data acquisition in flight, having on board a sophisticated avionic system.
In order to reduce the on ground processing time, a particular attention has been put to the filtering techniques performed at different steps of the process.
From this point of view the combat aircraft can give several problems if compared with transport aircraft, because of the high variability in the flight parameters values.
The advantages of this "individual" monitoring system are evidently stated and they can be briefly listed below:

- optimization of the fleet life management
- Economical benefits, coming from the knowledge of the aircraft service usage, which generally is lighter than the design required one (producing the design spectrum).
  So, it can be avoided to ground the aircraft for scheduled inspections at intervals defined in terms of flight hours, being convenient to inspect at reaching of the planned residual lives. Moreover, it has been noted that the in service utilization often goes out of the design life, relying on the large scatter factor value for fighters aircraft and on the fail safe properties for civil ones.
  In these cases a monitoring system can be considered of primary importance.
- Flight safety, because the system is analyzing all the components from which the safety is depending.

5. REFERENCES

1. J.B. De Jonge, D.J. Spiekhout, Assessment of Service Loads experience using aids - recorded data, Darmstadt, 11/13 May '77.
2. E.Bertolina, Flight Parameters Recording For Safety Monitoring And Investigations. AGARD CONFERENCE PROCEEDINGS No 347.

FLIGHT PARAMETERS LIST FOR STRUCTURAL MONITORING

- Wing	δ_{fl} δ_{sl} p $\dot{p}$ q α_t δ_{sp} N_z
- Tailplane (horiz.)	α_{tr} $\delta_{HT\ left,rig.}$ β p $\dot{p}$
- Vertical tail	p $\dot{p}$ β $\delta_{rud.}$ N_y r
- Rear fuselage	(L_{HT}) (L_{VT}) N_z N_y q
- Centre fuselage	N_z N_y α_t β q (L_{HT}) (L_{VT}) Sink sp. $\delta_{airbrakes}$
- Front fuselage	N_z N_y α_t q r
- Pylons	N_z N_y α_t p $\dot{p}$ q r

and additionally

Λ Mach Alt. Fuel Stores Xcg W

Fig. 1

LOADS BLOCK PROGRAM

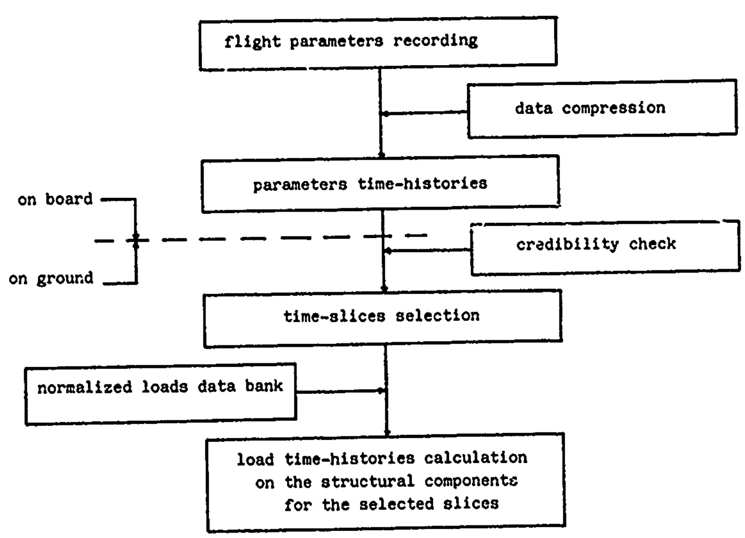


Fig. 2

FATIGUE BLOCK PROGRAM

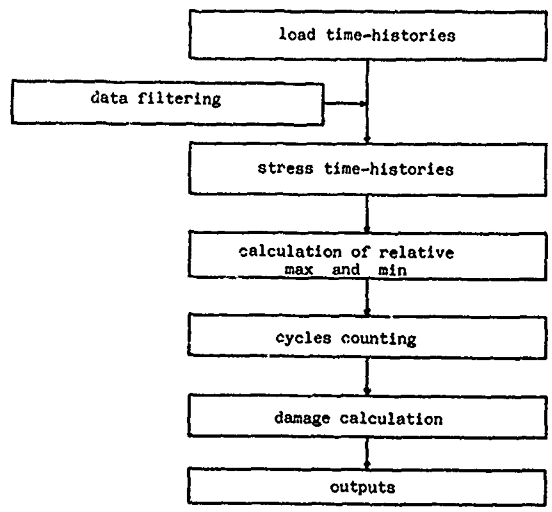


Fig. 3

DATA FILTERING FOR STRESS TIME-HISTORIES

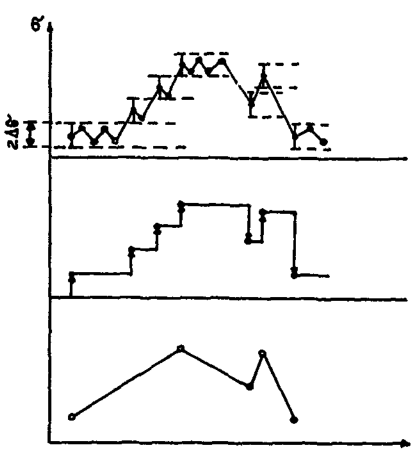


Fig. 4

CYCLES CONVERSION

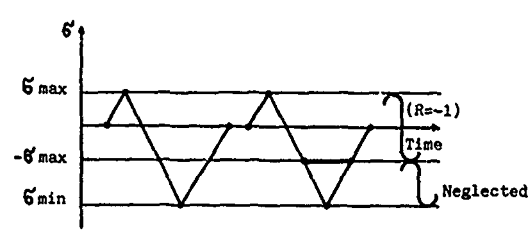


Fig. 5

AIRCRAFT DATA BANK ORGANIZATION

DATA BANK VECTOR FOR A/C IDENTIFICATION NUMBER 7001

N	COMPONENT	SERIAL NUMBER	ACTUAL FHS	S.F. LIFE	CUMUL. DAMAGE	LOAD/STRESS TR. RIF1	RIF2	S - N CURVES COEFFICIENTS (7 X 1 COMPONENT) A1	A2	A3	A4	A5	A6	A7
1	FRAME X 8000 S	1002.	10.25	0.40E+04	0.210-04	0.120E-01	0.012E-01	-3.665	0.233	2.155	-12.122	0.312	1.322	0.10E+08
2	FRAME X 8000 D	1013.	10.25	0.40E+04	0.230-04	0.120E-01	0.012E-01	-3.665	0.233	2.155	-12.122	0.312	1.322	0.10F+08
3	FRAME X 12737 S	3005.	10.25	0.40E+04	0.200-04	0.368E-01	0.000E-01	-2.998	0.243	2.985	-11.133	0.355	1.252	0.10E+07
4	FRAME X 12737 D	3061.	10.25	0.40E+04	0.200-04	0.368E-01	0.000E-01	-2.998	0.243	2.985	-11.133	0.355	1.252	0.10E+07
5	FRAME X 9110 S	2075.	10.25	0.40E+04	0.250-04	0.889E-01	0.115E-01	-3.991	0.385	3.752	-12.324	0.386	1.435	0.10E+06
6	FRAME X 9110 D	2088.	10.25	0.40E+04	0.280-04	0.889E-01	0.115E-01	-3.991	0.385	3.752	-12.324	0.386	1.435	0.10E+06
7	TAIL. TORQ. BOX S	8071.	10.25	0.40E+04	0.220-04	0.274E-01	0.000E-01	-3.677	0.488	2.621	-11.989	0.300	1.002	0.10E+07
8	TAIL. TORQ. BOX D	8081.	10.25	0.40E+04	0.230-04	0.274E-01	0.000E-01	-3.677	0.488	2.621	-11.989	0.300	1.002	0.10E+07
9	TAIL. SP. ROOT S	8111.	10.25	0.40E+04	0.300-04	0.108E-01	0.000E-01	-4.005	0.466	2.355	-9.122	0.463	1.989	0.50E+07
10	TAIL. SP. ROOT D	8111.	10.25	0.40F+04	0.310-04	0.108E-01	0.000E-01	-4.005	0.466	2.355	-9.122	0.463	1.989	0.50E+07
11	REAR FIN ATTACH. S	8121.	10.25	0.40E+04	0.300-04	0.775E-01	0.000E-01	-3.677	0.488	2.621	-11.989	0.300	1.002	0.10E+07
12	REAR FIN ATTACH. D	7125.	10.25	0.40E+04	0.310-04	0.775E-01	0.000E-01	-3.677	0.488	2.621	-11.989	0.300	1.002	0.10F+07
13	RIB 10/10/Y320 S	1005.	10.25	0.40E+04	0.210-04	0.313E-01	0.000E-01	-3.704	0.337	2.152	-10.100	0.374	1.128	0.10F+07
14	RIB Y10/10/320 D	1015.	10.25	0.40E+04	0.210-04	0.313E-01	0.000E-01	-3.704	0.337	2.152	-10.100	0.374	1.128	0.10E+07
15	S/T TR. BOX Y909 S	1025.	10.25	0.40E+04	0.260-04	0.409E-01	0.000E-01	-3.704	0.337	2.152	-10.100	0.374	1.128	0.10E+07
16	S/T TR. BOX Y909 D	1075.	10.25	0.40E+04	0.270-04	0.409E-01	0.000E-01	-3.704	0.337	2.152	-10.100	0.374	1.128	0.10E+07
17	I/B PYLON AREA S	1102.	10.25	0.40E+04	0.270-04	0.741E-01	0.004E-01	-3.704	0.337	2.152	-10.100	0.374	1.128	0.10E+07
18	I/B PYLON AREA D	1133.	10.25	0.40E+04	0.280-04	0.741E-01	0.004E-01	-3.704	0.337	2.152	-10.100	0.374	1.128	0.10E+07
19	WING FRONT SPAR S	1902.	10.25	0.40E+03	0.240-04	0.650E-01	0.010E-01	-3.704	0.337	2.152	-10.100	0.374	1.128	0.10E+07
20	WING FRONT SPAR D	1911.	10.25	0.40E+03	0.250-04	0.850E-01	0.010E-01	-3.704	0.337	2.152	-10.100	0.374	1.128	0.10E+07
21	DUMMY COMPONENT		10.25	0.40E+04	0.190-04	0.615E-01	0.000E-01	-5.297	0.100	2.110	-10.490	0.100	1.549	0.10E+07

Fig. 6

FUL OUTPUT

AIRCRAFT IDENTIFICATION NUMBER 7001

N	STRUCTURAL COMPONENT	SER. NUM.	SAFE FAT. LIFE (EQ.FHS)	FLIGHT HOURS (FHS)	EQUIVALENT FLIGHT HRS (EQ.FHS)	RESIDUAL LIFE (EQ.FHS)	CUMULATE DAMAGE	IM (EQ.FHS/FHS)
1	FRAME X 8000 S	1002.	4000.	10.25	0.85	3999.15	.21250-04	0.083
2	FRAME X 8000 D	1013.	4000.	10.25	0.92	3999.08	.23000-04	0.090
3	FRAME X 12737 S	3005.	4000.	10.25	0.79	3999.21	.19750-04	0.077
4	FRAME X 12737 D	3061.	4000.	10.25	0.80	3999.20	.20000-04	0.078
5	FRAME X 9110 S	2075.	4000.	10.25	0.99	3999.01	.24750-04	0.097
6	FRAME X 9110 D	2088.	4000.	10.25	1.12	3998.88	.28000-04	0.109
7	TAIL. TORQ. BOX S	8071.	4000.	10.25	0.88	3999.12	.22000-04	0.086
8	TAIL. TORQ. BOX D	8381.	4000.	10.25	0.91	3999.09	.22750-04	0.089
9	TAIL. SP. ROOT S	8111.	4000.	10.25	1.20	3998.80	.30000-04	0.117
10	TAIL. SP. ROOT D	8121.	4000.	10.25	1.23	3998.77	.30750-04	0.120
11	REAR FIN ATTACH. S	7125.	4000.	10.25	1.21	3998.79	.30250-04	0.118
12	REAR FIN ATTACH. D	7115.	4000.	10.25	1.25	3998.75	.31250-04	0.122
13	RIB 10/10/Y320 S	1005.	4000.	10.25	0.83	3999.17	.20750-04	0.081
14	RIB Y10/10/320 D	1015.	4000.	10.25	0.84	3999.16	.21000-04	0.082
15	S/T TR. BOX Y909 S	1025.	4000.	10.25	1.02	3998.98	.25500-04	0.100
16	S/T TR. BOX Y909 D	1075.	4000.	10.25	1.07	3998.93	.26750-04	0.104
17	I/B PYLON AREA S	1102.	4000.	10.25	1.08	3998.92	.27000-04	0.105
18	I/B PYLON AREA D	1133.	4000.	10.25	1.10	3999.89	.27500-04	0.107
19	WING FRONT SPAR S	1902.	4000.	10.25	0.97	3999.03	.24250-04	0.095
20	WING FRONT SPAR D	1911.	4000.	10.25	0.99	3999.01	.24750-04	0.097
21	DUMMY COMPONENT		4000.	10.25	0.76	3999.24	.15000-04	0.074

Fig. 7

SYNTHETIC OUTPUT

```
AIRCRAFT IDENTIFICATION NUMBER  7001

          DATE OF LAST FLIGHT            15/02/84
          TIME OF LAST FLIGHT             0.75   FHS
          A/C TOTAL FLOWN HOURS          10.25   FHS

          DATE OF A.G.S. PROCESSING      16/02/84
* * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * *
*                                                               *
*  MINIMUM RESIDUAL LIFE COMPONENT :             DUMMY          *
*          REAR FIN ATTACH. D 7115               COMPONENT      *
*                                                               *
*  RESIDUAL LIFE (EQ.FHS)        3998.747         3999.238      *
*                                                               *
*  EQ.FHS / FHS                     0.122            0.074      *
*                                                               *
* * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * *
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USAF APPROACH TO AIRBORNE STRUCTURAL DATA RECORDING
(Airborne Data Acquisition Multifunction System)
(ADAMS)

by
Ray J. Veldman
Aeronautical Engineer
Force Management Group
Aeronautical Systems Division
ASD/ENFSL, Wright-Patterson Air Force Base OH 45433

INTRODUCTION

The objective of this paper is to familiarize the reader with our current approach to structural recording and give some insight into our considerations for future instrumentation systems. For over forty years the USAF has used airborne instrumentation and recording systems to collect data that describe the structural loading environment that aircraft experience in their normal operational usage. These data are used for two purposes: 1) as a definition of the operational environment of the fleet of aircraft instrumented and its resultant impact on the design service life and 2) as design criteria for future aircraft of the same category. A multitude of instrumentation packages have been used to accomplish this task with varying degrees of success but the end result has been that in almost all cases, sufficient data was collected to accomplish the objectives of the recording program. For the last fifteen years magnetic tape digital recorders collecting data continuously using fixed sampling rates have primarily been used. Although these were excellent systems when initially introduced into the inventory, Air Force supportability of non-mission essential equipment and technology advancements have caused the continued use of such systems to be inappropriate. The current state of the art of microprocessor technology lends itself to the development of airborne recording systems capable of on-board processing and data compression with solid state data storage. Such systems will reduce supportability requirements drastically because of increased reliability inherent to solid state electronics while providing tremendous processing and self diagnostic capability heretofore unachievable. These microprocessor based systems which record structural operational information are designated within the Air Force as the ADAMS, Airborne Data Acquisition Multifunction System.

BACKGROUND

The requirement for airborne data recording is established in the Aircraft Structural Integrity Program (ASIP) as defined in AF Regulation 80-13 dated 16 July 1976. MIL-STD-1530A dated 11 December 1975 describes the implementation of ASIP for all USAF aircraft. Figure 1 is a table from this document and describes the tasks required to accomplish the ASIP. Task V, Force Management, illustrates the requirement for two types of airborne recording; the Loads Environment Spectra Survey (L/ESS) and the Individual Aircraft Tracking Program (IAT). According to MIL-STD-1530, the objective of the L/ESS is to obtain time history records of those parameters necessary to define the actual stress spectra for the critical areas of the airframe. The objective of the IAT is to predict the potential flaw growth in critical areas of each airframe that is keyed to damage growth limits of MIL-A-83444, inspection times, and economic repair times. L/ESS requires the instrumentation of a percentage (10-20%) of the fleet of aircraft with recording systems collecting multiparameter data such as CG load factors, angular rates, control surface positions, strains configuration, and events. Conversely, the IAT is accomplished on each flight of every aircraft and involves either manual data recording, flight logs, for transport/bomber aircraft or counting accelerometers/mechanical strain recorders for fighter aircraft. The concept is relatively simple. The L/ESS provides statistically average loading spectra for the aircraft fleet for all normal operating conditions (configuration, GW, CG, altitude, airspeed, etc) and the IAT defines the operating conditions experienced on an individual aircraft flight basis. The requirements for recording hardware/software used to accomplish these programs are what led to the concept of ADAMS and a discussion of the ADAMS system is the purpose of this paper.

DISCUSSION

For years airborne operational magnetic tape data recording has been plagued with many problems and constraints which have caused low valid data yield. The problems were not caused by bad recorder design but rather inherent limitations associated with magnetic tape recording and the inborne constraints associated with non-flight essential airborne avionics. Some of the more obvious problems affecting such a system are as follows:

1. Mechanical equipment within magnetic tape cartridges including the tape itself have unidentifiable but finite life due to wear. Conversely close tolerance on such parts is necessary for proper system operation.

2. Extremely high data tape packing densities are needed to achieve the required record duration. Such packing densities cause tape/tape head alignment to be extremely critical which results in considerable reading errors.

3. Maintaining a tape cartridge pipeline between the operational unit and the remote data transcribing and processing facility is a difficult logistics task.

4. There is the inability to self test the system other than for continuity. Most recorder system problems are not identified until the tape cartridge is transcribed at the central facility at Oklahoma City Air Logistics Center (OC-ALC). This ofren is three months after the tape is removed from the aircraft and all data recorded in this interim is invalid.

EDITOR'S NOTE

This final paper was first presented in a pilot paper version to the Operational Loads Data Sub-Committee at their meeting in Toronto, Canada. Mr Veldman gave a verbal update covering recent developments on the system described during Session IV of the Specialists Meeting – see relevant paragraphs in the Summary Record.

The obvious limitation of the system is the regular need for maintenance support even though it is non-flight essential equipment. This constraint will rightfully remain because the mission of the Air Force is to keep its weapon systems operationally ready and not to maintain structural monitoring systems. Maintenance on these systems remains low priority.

Recognizing the tremendous advancements in the state of the art of microtechnology, it was obvious that a microprocessor based solid state data collection system could eliminate many of the problems associated with previous systems while at the same time drastically reduce many of the supportability requirements that historically could not be satisfied. For some time this office had been investigating the state of the art of microprocessor technology for application to structural recording. We also had been monitoring an A-10 prototype installation of a microprocessor based engine monitoring system which showed great promise. Working with the B-1 System Program Office (SPO), we initiated the development of a new structural recording microprocessor based system. Rockwell International, the B-1 Prime Contractor, through an Advance Change Study Notice (ACSN) was charged with developing this system for the B-1 aircraft. Rockwell conducted their own source selection, evaluating proposals from several vendors and in July 1982 submitted a proposal to the B-1 SPO for the B-1 recording system. Along with the B-1 decision to use a microprocessor based structural recording system, has come a general Air Force trend toward such systems on all future Air Force aircraft. Many design features of the B-1 microprocessor system will be applicable to other aircraft.

A microprocessor based recording system eliminates most of the problems identified earlier. It is self diagnostic simply because the airborne microprocessor is programmed to interpret the validity of the data it is recording and "flag" problems as they occur. The entire system is solid state including the data storage eliminating the unreliable mechanical components inherent in magnetic tape recorders. Tape skewing and high packing densities are obviously eliminated.

Mini computers have had great processing capability for some time as evidenced by the explosive solid state time piece and electronic game market. The problem with the application of total solid state electronics to a mass storage recording systemhave been the data storage medium. The tremendous advancements in memory types, chip size, and design has only recently provided the capability for sufficient solid state storage capacity.

One of the primary considerations in the definition of the B-1 ADAMS was that of the functions it should include. USAF aircraft have many requirements for airborne data recording including structural or ASIP, ENSIP (Engine Structural Integrity Program), crash, engine performance, engine diagnostics, etc. Considerations in determining which functions to combine included solid state memory size, physical recorder size, who are the data users, data compression techniques associated with each function, common parameters, etc. The decision was made to include airframe structural recording, engine structural recording and the crash recorder option in the ADAMS system. Since the B-1 aircraft has no crash recorder requirement, this function was not included in the recording system requirements but inherently remains an option in the system design. The structural recording system philosophy included instrumenting ten percent of the aircraft for L/ESS and all aircraft for IAT. For hardware cost and aircraft configuration standardization purposes it was decided that a common system in each aircraft was the most viable approach. Therefore, the airborne microprocessor software would determine if the common hardware would sample and record L/ESS and IAT parameters or IAT parameters alone. For convenience this dual function could be chosen on any aircraft any time by merely swicching to one or the other software routines. Additionally, the L/ESS aircraft would include the six ENSIP parameters monitored on one of the four engines. This decision was somewhat obvious because of the close correlation between the L/ESS and ENSIP programs which both represent statistical descriptions of airframe or engine environments.

The following list depicts the B-1 parameters for L/ESS, IAT, and ENSIP.

*LH HORIZ STAB SPINDLE STRESS	*GROSS WT
RH HORIZ STAB SPINDLE STRESS	*CG
*LH CAP SPINDLE BASE STRESS	TOTAL FUEL QUANT
WING SWEEP ACT STRESS	NOSE GR SQUAT SW
*WING OUTER PANEL STRESS	MAIN GR SQUAT SW
*FUSELAGE FOREBODY STRESS	
	OAT
PITCH RATE	*MACH NO
ROLL RATE	*PRES ALT
YAW RATE	AIRSPEED
*VERT ACCEL	**ENG INLET TEMP
LAT ACCEL	$**N_1$
LONG ACCEL	$**N_2$
	**PLA
RH HORIZ STAB POS	**BLADE TEMP
LH HORIZ STAB POS	**COMP STATIC PRESS
RH INBOARD SPO POS	
LH INBOARD SPO POS	
UPPER RUDDER POS	
*RH FLAP POS	
*WING SWEEP POS	

NOTE: * - IAT
** - ENSIP

The B-1 system will consist of three pieces of hardware - the Structural Data Collector (SDC), Structural Data Extractor (SDE), and Structural Data Transcriber (SDT). The SDC is the airborne microprocessor/recorder and the SDE and SDT are ground support equipment. The airborne unit will have 256K Bytes mass storage memory to record significant data from L/ESS, IAT, and ENSIP functions. It will be

programmed to compress t input data in order to record only those samples which are defined as structurally significant. Several techniques for data compression have been developed over the years including peak counting, threshold or window levels, key parameters with coincident values, etc. While the method or methods to be used for the B-1 are not finalized at this time, the 256K Byte memory requirement is considered adequate to accommodate whatever method chosen. This size memory should allow for at least 20 flight hours of L/ESS/ENSIP data and 200 flight hours of IAT data before a memory Jump is required. The SDE will be a hand held ground unit which provides the ability to extract data from several airborne units and store the information in its own solid state memory. This unit will be field level support equipment at least one available at each base depending on the number of aircraft at the base. It will also display any system discrepancies or failures diagnosed by the airborne system and alert field personnel that maintenance action is required. For the B-1 this is somewhat of a redundant feature because the ADAMS will have a Central Integrated Test System (CITS) interface which will display an ADAMS system failure in real time. The SDT will be a ground based shop level system capable of transcribing multiple extractions on a magnetic tape or floppy disk storage medium to be sent to the agency responsible for data analysis. This system will also allow those personnel to "quick look" and operate on any data recorded by the SDC to diagnose possible system problems.

Much of the technical design detail of the B-1 ADAMS system has been defined at this time but some remains an open consideration. A discussion of these design details would be premature and inappropriate for this paper, but could be the subject of a follow-on paper.

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2. MIL-STD-1530A, 11 December 1975, Aircraft Structural Integrity Program, Airplane Requirements.

USAF Aircraft structural integrity programs tasks.

TASK I	TASK II	TASK III	TASK IV	TASK V
DESIGN INFORMATION	DESIGN ANALYSES AND DEVELOPMENT TESTS	FULL SCALE TESTING	FORCE MANAGEMENT DATA PACKAGE	FORCE MANAGEMENT
ASIP MASTER PLAN	MATERIALS AND JOINT ALLOWABLES	STATIC TESTS	FINAL ANALYSES	LOADS/ENVIRONMENT SPECTRA SURVEY
STRUCTURAL DESIGN CRITERIA	LOAD ANALYSIS	DURABILITY TESTS	STRENGTH SUMMARY	INDIVIDUAL AIRPLANE TRACKING DATA
DAMAGE TOLERANCE & DURABILITY CONTROL PLANS	DESIGN SERVICE LOADS SPECTRA	DAMAGE TOLERANCE TESTS	FORCE STRUCTURAL MAINTENANCE PLAN	INDIVIDUAL AIRPLANE MAINTENANCE TIMES
SELECTION OF MAT'LS, PROCESSES & JOINING METHODS	DESIGN CHEMICAL/ THERMAL ENVIRONMENT SPECTRA	FLIGHT & GROUND OPERATIONS TESTS	LOADS/ENVIRONMENT SPECTRA SURVEY	STRUCTURAL MAINTENANCE RECORDS
DESIGN SERVICE LIFE AND DESIGN USAGE	STRESS ANALYSIS	SONIC TESTS	INDIVIDUAL AIRPLANE TRACKING PROGRAM	
	DAMAGE TOLERANCE ANALYSIS	FLIGHT VIBRATION TESTS		
	DURABILITY ANALYSIS	FLUTTER TESTS		
	SONIC ANALYSIS	INTERPRETATION & EVALUATION OF TEST RESULTS		
	VIBRATION ANALYSIS			
	FLUTTER ANALYSIS			
	NUCLEAR WEAPONS EFFECTS ANALYSIS			
	NON-NUCLEAR WEAPONS EFFECTS ANALYSIS			
	DESIGN DEVELOPMENT TESTS			

Figure 1

SESSION IV - FUTURE SYSTEMS

SUMMARY RECORD

by
R G Loewy
School of Engineering,
Rensselaer Polytechnic Institute,
Troy, NY, USA

The first paper in the session (paper No 16) covered questions of instrument selection, numbers of channels, sampling rates, aircraft areas of emphasis, calibration and data collection concepts. The presenter noted good agreement between results of finite element analysis and regression analysis of stresses measured in ground tests. As an example of the data being taken, he showed time histories in which wing bending, taileron and fin data appeared for various sampling rates. In reply to a question on how fatigue life information on the entire Tornado fleet was to be inferred from a limited number of fully instrumented aircraft, he noted that the issue had been addressed, but could not be fully resolved until more experience had been accumulated; however, use would be made of the vertical acceleration recorders fitted to all aircraft.

The second presenter (paper No 17) noted that most French fighters, attack aircraft and trainers carried acceleration counting systems recording exceedences of various g thresholds, 100% of the Mirage F-1 and Alpha-Jet fleet being fitted, but only 10% of, for example, Mirage III and 5 aircaft. He described the criteria used to develop a new system of operational loads information acquisition, including low maintenance requirements, easy interchange between types, ease of data management and provision of loads data applicable beyond the aircraft type on which it was acquired. He noted that still more advanced systems should include large memory, quick scanning capability, deferred processing and use of digital data bus techniques.

The third paper (paper No 18) referred to experience gained with the maintenance recorder in the Italian Air Force Tornado fleet. Proven structural life is considered as that life demonstrated in tests, rather than life calculated using assumptions such as Miner's Rule. Flight parameters and processing routines were described, including data-filtering logic for stress-time histories (based on threshold changes of 5 Newtons/sq mm). The presenter showed sample data including percentage structural component life on a flight hours basis, and closed by noting the complexity of systems which are, conceptually, relatively simple, together with the need for a greater degree of automation in data handling.

In general discussion, the question of cooperation and correlation of results on the two Tornado programmes described was raised. So far, no cooperative arrangements had been set up. An Italian Air Force questioner referred to the number of flight strain surveys that had been carried out on Tornado, and asked whether operational loads and strain survey data could not be correlated, perhaps operational loads surveys should be carried out before fatigue qualification testing. In reply, the UK view was that although early flight strain data under defined flight test conditions were fully accepted, loads encountered in service operations were often different from those specified as design loads and confirmed in strain surveys. If possible, there should be, and often were, two fatigue tests, one early in the development cycle and one considerably later, after operational loads data had been acquired. On this point, one speaker felt that a second fatigue test was often required because the first test was too abbreviated or there had been significant structural changes, rather than because the load spectrum had been redefined.

At this stage in the proceedings, the Meeting Chairman invited Mr R Veldman to give an update of the pilot paper given at the Toronto meeting of the Panel. This paper describes the USAF multi-function aircraft data acquisition system currently under development; an updated version is included with the Specialist Meeting proceedings as paper No 19. The author observed that the reliability of tape-recorder based systems had been a major problem in US programmes. The new system utilised on-board micro processors and solid-state data storage, and was expected to give a 'state of the art' improvement. Delivery of production equipments was expected in mid-1984 and the system was intended for both the B-1B and T-46A programmes.

In response to two questioners both asking if the 'write cycle' of the EEPROM data storage did not require larger memory than other equivalent systems, the presenter replied that this problem was avoided by parallel operation. Moreover, data is never erased; locations are filled progressively, data is extracted and, on further use, new data is overwritten over old. In this mode, except at very low temperatures, data system speed is adequate.

ROUND TABLE DISCUSSION

Meeting Chairman Bright then convened a Round Table Discussion, inviting the Session Chairman, Messrs de Jonge, Maxwell, Chesta and Veldman to join him. Opening the discussion, he asked the audience to think in terms of current requirements for operational loads data (systems, format and so on) on the one hand, and systems likely to be needed for the future, say the 1990s, on the other. Agile aircraft, exploiting the full potential of ACT, digital computing and composite structures, were likely to pose quite new problems for loads data acquisition. Were we convinced as to the continuing need for this kind of information; did we continue to pursue the two-tier concept, in which only a small fraction of an aircraft fleet is fully instrumented, with limited instrumentation common to the entire fleet?

Mr de Jonge said that he thought the Chairman's questions were well-posed. He then raised the issue of data scatter, when operation of one aircraft type extends over a long period of time, with the aircraft rotated between different squadrons and pilots. His experience with the F-104 under these circumstances suggested that scatter is statistically 'smoothed-out', perhaps implying that small samples are adequate. Mr Maxwell observed that squadron roles can differ substantially, with noticeable effect on airframe loads; while too much data can swamp ground analysis, carefully selected and processed data can be used to identify usage patterns and loads effects, any exceedences of the design envelope and the basic causes of high fatigue damage.

Mr Graham observed that it is necessary to define the structural damage under consideration rather narrowly and the statistical distribution must be known before 'scatter' as a concept has any real meaning. He noted that the F-5 fleet data shows a factor of 4 in the structural life remaining for aircraft with 10-12 years of operations, including rotation of aircraft between squadrons. Messrs Maxwell, Bright and de Jonge then agreed that the exact nature of usage patterns, eg. normal peace-time fighter squadron operations vs. training, had a major influence on loads spectra; for scatter among the same type to smooth out, the aircraft must all be consistently operated in the same way.

Netherlands F-104 scatter was of the order of 2. For comparison, Mr Neunaber said that German Air Force F-104 scatter was reported as 2.3 (by normal distribution, 10-90 percentiles). If aircraft are not rotated between squadrons, the scatter is 3-4. It was necessary to be precise in defining 'scatter', otherwise discussion might be at cross-purposes. Returning to discussion of the 2.3 factor mentioned above, it was thought that its causes included differences in control responsiveness between individual aircraft; many pilots may take advantage of such differences between aircraft in their handling; possibly scatter will be reduced over a larger fleet life-time. Wg Cdr Bright noted that RAF experience, like French Air Force experience, tended to show that operational load severity increases with time. M Baranes said that he had seen scatter factors for the same mission as high as 5.

Mr Krauss asked how evolutionary changes in a particular aircraft type's structure, which often occur during its operational life time (eg due to retrofits) are taken into account in monitoring fatigue life. Wg Cdr Bright replied that such differences must be accounted for as they occur, citing possible differences in F-4J aircraft about to be introduced to RAF service, following operation of other versions of F-4.

Mr Hacklinger referred to basic differences between design conditions and those actually encountered in operations, and said that these needed to be considered from two view points. Firstly, as an uncertainty in performing 'on-condition' maintenance, where the actual history and what is likely to be encountered must both be known, and secondly, what must be known to design better new aircraft. On the second point, there was a dearth of information available during the design phase as to how a new aircraft type would, in fact, be flown. Professor Loewy asked if advanced simulators might not be useful in this context. Mr Maxwell doubted their value, quoting Kestrel/Harrier experience which did not anticipate combat use of vectored thrust. Wg Cdr Bright noted that, in any event, operational loads data was required to 'close the loop'. Mr Krauss agreed, commenting that in a simulator pilots can fly only a 'nominal' aircraft.

Wg Cdr Bright asked the assembly for their views on ground versus airborne data processing. Mrs Holford replied that she would like to address the topic from two view-points. On the one hand, she questioned the advisability of using the same processors, sampling and filter logic, etc for both loads survey and in fatigue life tracking, since then both activities might miss the same, possibly important, effect. On the other hand, she was convinced that highly instrumented loads data acquisition aircraft could incorporate certain automatic flight control systems which would make them non-representative of the fleet at large. Addressing the first point, Mr Veldman observed that the new USAF system he described had four different programmable logic algorithms for the data processing functions.

The Chairman asked for consideration of the applicability of current loads data for emerging agile aircraft designs. Mr Culp stated that General Dynamics' testing of early F-16 models provided a very large flight data base which had proved useful in C and D model developments, and for the XL version with a composite wing. He note that these are moderately agile aircraft. The F-16 crash recorder has a large memory (4-5 megabytes) and GD uses it to predict loads, obtain stability and control information, etc. Mr Krauss commented that lack of knowledge of aerodynamic stability derivatives in extreme manoeuvres makes load prediction doubly difficult in design of agile aircraft. Mr Hacklinger added that it is important not to rely exclusively upon existing data in designing such aircraft. He noted, for example, that the extensive use of composites makes design less fatigue-critical and that manoeuvre limiting systems are being used more frequently. In consequence, new balances will have to be struck between manoeuvrability and strength.

Mr Johnston observed that an overload warning system is working quite well on F-15 aircraft. Simulations of automatic load-limiter systems, as distinct from g-limiter systems, had been run on both F-15 and F-18 aircraft. He believed that all advanced, high performance aircraft will make use of such systems in the future. Wg Cdr Bright noted that for aircraft with such systems on board, simply recording that system's activity would constitute operational loads data. Mr Maxwell added that he would wish to have simultaneous control position information.

CLOSURE

Following the discussion, the Chairman thanked the authors, session chairmen and assembled participants for their contributions, which had produced a highly successful specialists meeting on Operational Loads Data. He then declared the session adjourned.

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14. Abstract

The aim of this Meeting was to examine the general principles followed in acquiring data on operational loads sustained by NATO aircraft, both fixed and rotary wing, in normal service and to compare current and developing loads data acquisition programmes. The techniques used and data obtained from work on a variety of aircraft types were reported; microprocessor-based systems were seen as providing the essential technology for major advances in this field. Although the main emphasis was on fatigue loads, the acquisition of design loads data was also discussed, together with the validity of current data bases for loads calculations on high-agility designs representative of the next generation of combat aircraft.

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